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USAAEFA PROJECT NO. 79-24

VALIDATION FLIGHT TEST OF UH-60A FOR ROTORCRAFT SYSTEMS INTEGRATION SIMULATOR (RSIS)

FINAL REPORT

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UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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The United States Army Aviation Engineering Flight Activity (USAAEFA) conducted flight tests of the UH-60A helicopter for the Aeronmechanics Laboratory (AL) of the US Army Aviation Research and Technology Laboratories to obtain data for validation of a simulator. A test program of sixty-nine flights totaling 97.4 test hours was flown to provide AL with extensive handling qualities data. In addition to standard handling qualities tests, special system identification maneuvers were flown. Summary results of those tests are contained in this report.

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DEPARTMENT OF THE ARMY
HQ, US ARMY AVIATION RESEARCH AND DEVELOPMENT COMMAND
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DRDAV-D

SUBJECT: Directorate for Development and Qualification Position on the Final Report of USAAEFA Project No. 79-24, Validation Flight Test of the UH-60A for the Rotorcraft Systems Integration Simulator (RSIS)

SEE DISTRIBUTION

1. The purpose of this letter is to establish the Directorate for Development and Qualification position on the subject report. The reports documents in detail the acquired flight test data to be used by the Aeromechanics Laboratory (AL), US Army Aviation Research and Technology Laboratories (RTL), US Army Aviation Research and Development Command (AVRADCOM) in validating the RSIS degree-of-freedom moving base simulator. The report also provides qualitative remarks based on pilot observation and conclusions since the flight testing was conducted only to obtain quantitative control and response characteristics data from which to obtain basic aircraft stability derivatives for the RSIS validations of the fully qualified UH-60A.

2. This Directorate agrees with the report remarks and conclusions.

FOR THE COMMANDER:

CHARLES C. CRAWFORD, JR.
Director of Development
and Qualification

PREFACE

The flight test for simulator validation of the UH-60A Black Hawk was conducted by the United States Army Aviation Engineering Flight Activity (USAAEFA) at Edwards Air Force Base at the direction of the Aeromechanics Laboratory (AL) of the US Army Research and Technology Laboratories. USAAEFA wishes to acknowlegde the contributions of Raymond S. Hansen, aerospace engineer for AL. Mr. Hansen was instrumental in establishing the flight test matrix, defining the instrumentation requirements, obtaining leases for special rotor blade instrumentation, and acting as point of contact for financial matters. During the actual flight test, Mr. Hansen authorized modification to the test plan. The authors further wish to acknowledge the contributions of Joseph Piotrowski, a co-operative engineering student at Emory-Riddle University, Prescott, Arizona, who was working at USAAEFA during the test. Mr. Piotrowski monitored the telemetry station during test flights, reduced the data after the flights, and made preliminary plots for this report.

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INTRODUCTION

BACKGROUND

1. The Aeromechanics Laboratory (AL), of the US Army Aviation Research and Technology Laboratories (ARTL), US Army Aviation Research and Development Command (AVRADCOM), is developing a Rotorcraft Systems Integration Simulator (RSIS) to investigate flight control systems, augmentation systems, and displays that are being integrated into modern helicopters. The simulator will permit selective variation of parameters which determine the aircraft response, and affect the pilot's workload and ability to perform mission tasks. Such investigations will provide information concerning desired levels of flying and handling quality characteristics for the development of future Army helicopters, and allow evaluation of modifications to control system components and automatic flight control systems.

2. The value of a research simulator is largely measured by its ability to accurately simulate the control and response characteristics of an actual helicopter. Simulation validation is therefore required as early as possible in the development and use of RSIS. The helicopters used to perform this validation must represent current state-of-the-art aircraft which are expected to remain in the Army inventory for a significant length of time. The UH-60A fulfills this requirement, and was chosen for the validation test.

3. In October 1980, AVRADCOM requested that the United States Army Aviation Engineering Flight Activity (USAAEFA) perform validation flight tests on the UH-60A helicopter at Edwards AFB, California (ref 1, app A). The flight test matrix and instrumentation requirements were determined jointly by USAAEFA and AL. USAAEFA provided a test plan in December 1980 (ref 2).

TEST OBJECTIVE

4. The objective of this program was to generate and provide AL with flight test data necessary to define the control and response characteristics of the UH-60A. Significantly greater detail in instrumentation parameter scope and accuracy was required than is normal in a handling qualities program.

DESCRIPTION

5. The RSIS is a 6 degree-of-freedom, moving base simulator that incorporates advanced visual displays and is capable of simulating hovering and low speed nap-of-the-earth flight as well as high

speed flight. The simulation system consists of five major components : (1) an interchangeable cab including cockpit instruments, controls and displays; (2) a motion base which moves the interchangeable cab to duplicate aircraft motions; (3) a computer programmed with a mathematical model to control the motion of the cab and cockpit displays; (4) a visual system that generates the appropriate visual cues for the pilot; and (5) the pilot. The simulator also has the capability of generating aural and control force/feel cues for the pilot. The "end-to-end" simulation occurs when the pilot generates control inputs used in the mathematical equations of motion to drive the motion and visual systems.

6. The test helicopter, UH-60A US Army S/N 77-22716 (photo A), manufactured by Sikorsky Aircraft Division of United Technologies Corporation, is the third production Black Hawk. The UH-60A is a twin engine, single main rotor helicopter with fixed wheel-type landing gear. A movable horizontal stabilator is located on the lower portion of the tail rotor pylon. The main and tail rotor are both four-bladed with a capability of manual main rotor blade and tail pylon folding. The cross-beam tail rotor with composite blades is attached to the right side of the pylon. The tail rotor shaft is canted 20 degrees upward from the horizontal. Primary mission gross weight is 16,260 pounds and maximum alternate gross weight is 20,250 pounds. The UH-60A is powered by two General Electric (GE) T700-GE-700 turboshaft engines having an installed thermodynamic rating (30 minute) of 1553 shaft horsepower (shp) each at sea level, standard-day static conditions. The transmission is limited to 2828 shp. The aircraft also has an automatic flight control (AFSC) and a command instrument system (CIS). A more detailed description of the UH-60A is included in appendix B and additional descriptions can be found in the operator's manual (ref 3, app A) and the final report of USAAEFA Airworthiness and Flying Characteristics evaluation of the Black Hawk (ref 4).

TEST SCOPE

7. The major portion of the flight testing was conducted at Edwards AFB, California (elevation 2303 feet), with several flights conducted at Bakersfield, California (elevation 490 feet). Sixty-nine flights were made between 28 July 1981 and 6 May 1982 totaling 97.4 test hours. Except for rotor blade angle measurements, all test instrumentation was installed, calibrated, and maintained by USAAEFA personnel. Blade angle instrumentation was designed, installed, and initially calibrated by Sikorsky. The aircraft was maintained and flown by USAAEFA personnel. The flight crew consisted of two experimental test pilots and one

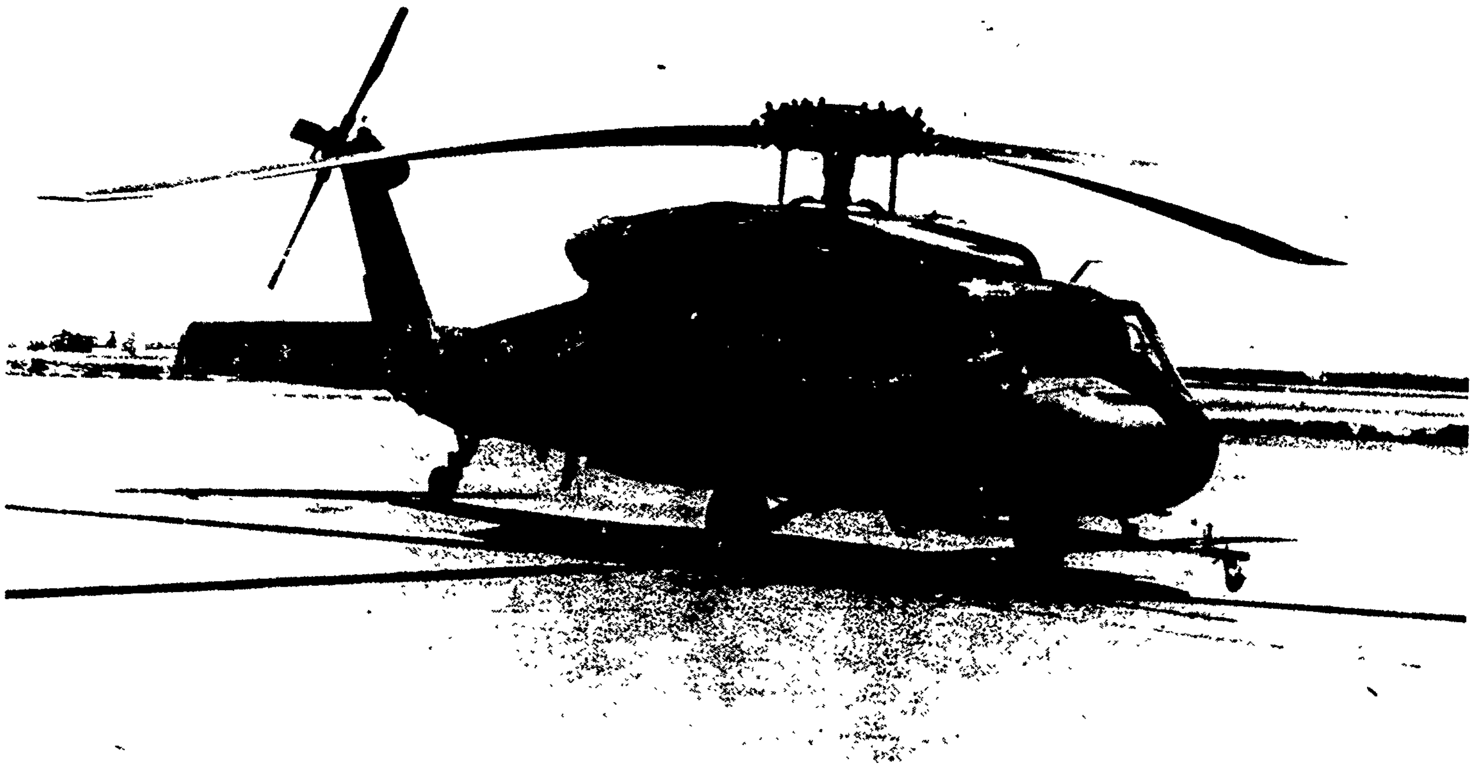


Photo A. UH-60A Black Hawk

flight test engineer. Flight limitations imposed by the operator's manual and the airworthiness release (ref 5) were observed at all times. Testing was in accordance with the test plan as modified in conjunction with AL, at the conditions shown in tables 1 through 3. Representative data are published in this report. Data not published are available through the USAAEFA technical library. Additionally, two flights (3.9 hours) were conducted under the direction of Systems Technology, Inc. (STI) of Palo Alto, California, a contractor of AL. The STI flights featured several pilots performing mission oriented tasks such as nap-of-the-earth, bob ups, quick stops, accelerations, etc. STI will use this data to independently evaluate RSIS end-to-end simulator performance.

TEST METHODOLOGY

8. Flight test data were obtained from test instrumentation displayed to the pilots and flight test engineer, and recorded on magnetic tape. Selected critical parameters were monitored on all dynamic maneuvering tests using telemetry. A detailed listing of test instrumentation is contained in appendix C. Variations on established flight test techniques (refs 6 and 7, app A) are detailed in appendix D. Test methods are also briefly described in the Results and Discussion section of this report. A Handling Qualities Rating Scale (HQRS) and Vibration Rating Scale (VRS) (figs. 1 and 2, app D) were used to augment pilot comments relative to handling qualities and vibration.

9. Dynamic control inputs were made using a control fixture. Timing of the dynamic inputs (particularly the System Identification inputs) was done with the aid of a cathode ray tube (CRT) display in the cockpit controlled by the engineer. A description of this system is contained in appendix C.

10. The successful completion of this test program was due, in part, to the close cooperation between USAAEFA and AL. Comments on the management of this program are contained in appendix E.

Table 1. General Test Conditions

The following conditions were held for all trim points throughout the test, except where alternate conditions are listed in tables 2 and 3.

Automatic Flight Control System Conditions

Pitch Bias Actuator (PBA) - Disabled and centered
Flight Path Stabilization (FPS) - Off
Trim System - Pilot's discretion
Stability Augmentation System (SAS) - Statics - On
- Dynamics - Off

Stabilator - Fixed according to the following schedule for specified $V_T/\sqrt{\theta}$
- Hover: 43° Trailing Edge Down (TED)
- $V_T/\sqrt{\theta}$ = 60 KTAS: 31° TED
- $V_T/\sqrt{\theta}$ = 100 KTAS: 8° TED
- $V_T/\sqrt{\theta}$ = 140 KTAS: 6° TED

- Programmed mode for static airspeed sweeps

Baseline Flight Conditions

Thrust Coefficient (C_T) = 68.5 x 10^{-4}, W/δ = 19,350 pounds at $N_R/\sqrt{\theta}$ = 100% (corresponds to 15,475 pounds at 5,000 feet)
Referred rotor speed, $N_R/\sqrt{\theta}$ = 100% = 257.9 RPM
Primary trim referred true airspeeds, $V_T/\sqrt{\theta}$ = 0, 60, 100, 140 knots
Longitudinal center of gravity = FS 351
Sideslip Angle = Zero
Out-of-ground effect
Level flight

Deviations from Baseline Conditions

Aft center of gravity, FS 359
Low rotor speed - $N_R/\sqrt{\theta}$ = 96% = 247.6 RPM
Primary trim speeds at low rotor speeds - $V_T/\sqrt{\theta}$ = 0, 57.6, 96, 134.4 knots
Mid C_T I = 80 x 10^{-4}, W/δ = 22,606 pounds at $N_R/\sqrt{\theta}$ = 100% (corresponds to 15,550 pounds at 10,000 feet)
Mid C_T II = 80 x 10^{-4}, W/δ = 22,606 pounds at $N_R/\sqrt{\theta}$ = 100% (corresponds to 18,900 pounds at 5,000 feet)
High C_T = 94.5 x 10^{-4}, W/δ = 26,845 pounds at $N_R/\sqrt{\theta}$ = 100% (corresponds to 18,465 pounds at 10,000 feet)
Very High C_T = 112 x 10^{-4}, W/δ = 31,649 pounds at $N_R/\sqrt{\theta}$ = 100% (corresponds to 17,869 pounds at 15,000 feet)

Table 2. Static Flight Test Conditions

Test	Trim Conditions[1]	Trim Referred True Airspeed $V_T/\sqrt{\theta}$ ~ knots	Remarks
Control Positions in Trimmed Forward Flight	Baseline	0 to V_H[2]	
	Aft cg		
	96% $N_R/\sqrt{\theta}$		
	Mid C_T I		
	Mid C_T II		
	High C_T	V_{min}[3] to V_H[2]	
	Very High C_T		
	Maximum C_T with ability to hover	0 to V_H	
	Lateral cg = BL + 2		
	Lateral cg = BL − 4		
	baseline ϕ = 0	note[4] to V_H[2]	Zero side force
	Longitudinal CG = FS 346	0, 60, 100, 140	$C_T = 67\times10^{-4}$, $N_R/\sqrt{\theta} = 100\%$
	Longitudinal CG = FS 346		$C_T = 67\times10^{-4}$, $N_R/\sqrt{\theta} = 96\%$
	Longitudinal CG = FS 346		$C_T = 80\times10^{-4}$, $N_R/\sqrt{\theta} = 100\%$
	Longitudinal CG = FS 352		$C_T = 67\times10^{-4}$, $N_R/\sqrt{\theta} = 100\%$
	Longitudinal CG = FS 352		$C_T = 67\times10^{-4}$, $N_R/\sqrt{\theta} = 96\%$
	Longitudinal CG = FS 356		$C_T = 67\times10^{-4}$, $N_R/\sqrt{\theta} = 100\%$
	Longitudinal CG = FS 356		$C_T = 67\times10^{-4}$, $N_R/\sqrt{\theta} = 96\%$
	Longitudinal CG = FS 360		$C_T = 67\times10^{-4}$, $N_R/\sqrt{\theta} = 100\%$
	Longitudinal CG = FS 360		$C_T = 67\times10^{-4}$, $N_R/\sqrt{\theta} = 96\%$
Control Positions in Climbs and Descents	Baseline		R/C = 0, maximum, 1/2 maximum
	96% $N_R/\sqrt{\theta}$	96	
	Aft cg	60, 100	R/D = maximum, 1/2 maximum
Low Speed	Baseline	0 to 40 forward, rearward, & sideward	100 foot wheel height
	Aft cg		
IGE Hover	Baseline	0	0, 5, 10, 25, 50, 75, 100, 150 foot wheel heights
Level Turns	Baseline	60, 100	30° & 45°, left and right bank angles
	Aft cg	100	
Descending Turns	Baseline	100, 140	1.5, 2.0, & 2.3 G in both directions
Static Longitudinal Stability	Baseline	60, 100, 140	Level flight
	Aft cg	60, 100	
	Mid C_T I	100	
	Mid C_T II	100	
	96% $N_R/\sqrt{\theta}$	96	
	Baseline Climbs/Descents	100	R/C = ±1500 ft/min
Lateral-Directional Stability	Baseline	60, 100, 140	Level flight
	Aft cg	60, 100	
	Mid C_T I	100	
	Mid C_T II	100	
	96% $N_R/\sqrt{\theta}$	96	
	Baseline Climbs/Descents	100	R/C = ±1500 ft/min
Rotor Speed Sweep	Baseline	0, 60, 100, 140	N_R = 96, 98, 100, 102%
	Mid C_T I	100	
Stabilator Sweep	Baseline	60, 100, 140	Maximum allowable stabilator travel
	Aft cg	100	

NOTES:

[1]Conditions other than those listed are baseline conditions (see table 1)
[2]V_H: Maximum level flight speed with test power available
[3]V_{min}: Minimum level flight speed with test power available
[4]Minimum level flight speed at which sideforce cues are apparent

Table 3. Dynamic Maneuver Test Conditions

Test	Trim Conditions[1]	Trim Referred True Airspeed $V_T/\sqrt{\theta}$ ~ knots	Remarks
Step Inputs (Longitudinal, Lateral, Directional, Collective)	Baseline	0, 60, 100, 140	1/4, 1/2, 3/4, and 1 inch, both directions
	Aft CG	0, 100	
	SAS ON		
Pulse Inputs (all controls)	Baseline	0, 60, 100, 140	1 inch, both directions
Doublets (Collective only)	Baseline	0, 60, 100, 140	Build up to 1 inch, both directions
	Aft CG	60, 100	
Doublets (Longitudinal, Lateral, Directional)	Baseline	0, 60, 100, 140	
	Aft CG	100	
SI Input (all controls)	Baseline	60, 100	
SI Input (Lateral only)	Aft CG	100	
Roll Reversals	Baseline	100, 140	ϕ = 30°, both directions
Sideslip Reversals	Baseline	100	β = 20°, both directions
Phugoid Excitement	Baseline, SAS ON	100, 140	10 knots fast and slow
Pushovers & Pullups	Baseline	100, 140	0 G, 2 G

NOTE:
[1]Conditions other than those listed are baseline conditions (see table 1)

RESULTS AND DISCUSSION

GENERAL

11. The AL requirement was to obtain simulator validation data for the basic unaugmented airframe and control system. Therefore, most of the functions of the AFCS were turned off. This results in a highly degraded configuration, and the data contained within this report cannot be considered representative of the flying qualities of the UH-60A in normal operating conditions. The unique nature of this program made specification compliance comparisons unnecessary. In particular, the following elements of the AFCS were degraded:

a. The automatically programmed stabilator is designed to optimize the aircraft pitch attitude for any flight condition. The programming function was turned off, and the stabilator was fixed in the position the program would normally have set for the aim airspeed at the baseline thrust coefficient (C_T).

b. The pitch bias actuator (PBA) is a variable length actuator in the longitudinal cyclic control system to assure a stable gradient of longitudinal cyclic versus airspeed. The PBA was disconnected and set to mid-length.

c. The stability augmentation system (SAS) provides three-axis rate damping and pseudo attitude hold. It was allowed to remain on for static tests, but was turned off for dynamic maneuvers.

d. The flight path stabilization (FPS) system is an attitude hold system that incorporates conditional capability for airspeed hold and turn coordination. It was turned off throughout the program.

12. The data were flown maintaining constant aim C_T using the referred gross weight (W/δ), referred main rotor speed ($N_R/\sqrt{\theta}$) method. Thus, altitude was increased as fuel was used, and main rotor speed decreased as temperature decreased. Trim conditions were flown at zero angle of sideslip.

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

13. Control positions in trimmed forward flight were evaluated in level flight and climbs and descents at the conditions shown in tables 1 and 2. Test results are presented in figures 1 through 11, appendix F. All data were flown at zero sideslip except that presented in figure 2. Airspeeds less than 40 knots calibrated airspeed KCAS were measured using the Marconi-Elliott Low Airspeed Sensing and Indicating Equipment (LASSIE).

14. At all conditions tested, increasing forward longitudinal cyclic was required for increased trim airspeed. Considerable nonlinearities in longitudinal cyclic position were noted at airspeeds less than 50 KCAS. Steady data at those speeds were considerably more difficult to obtain than at higher airspeeds. Pitch attitude varied from 2° nose up at hover to 8° nose down at V_H.

15. At all conditions tested, increasing right lateral cyclic was required with increasing airspeed to 130 KCAS. Above 130 KCAS, lateral cyclic control position remained generally constant. Total lateral cyclic movement between a hover and V_H was approximately 2 inches. Lateral cyclic position during the right lateral cg (BL +2) flight was 0.5 to 1 inch left of the baseline (fig. 5). At the left lateral cg (BL -4), lateral cyclic was 0.6 to 0.8 inches right of the baseline.

16. Increasing right directional control was required with increasing airspeed to approximately 90 KCAS. At airspeeds greater than 90 KCAS, directional control position remained relatively constant. Ball centered flight generally required about 0.5 inch additional right pedal compared to zero sideslip. Inherent sideslip was greatest at 67 KCAS (-4°) and varied linearly with increasing airspeed to 150 KCAS (-1°) (fig. 2).

17. Control positions in climbs and descents were evaluated as a function of rate of climb and descent from autorotational descent to Military Rated Power (MRP) climb at several constant airspeeds and at two cg configurations. At the mid cg configuration, (fig. 10), increasing amounts of forward cyclic control were required to maintain an equivalent rate of climb as the airspeed was increased. During descents at these conditions, the longitudinal cyclic position remained relatively constant; however, increasing amounts of right cyclic were required for increasing airspeed at constant rates of descent. A slight amount of increasing right pedal was required for increasing airspeed during climbs, and directional control position remained relatively constant for descents at the three test airspeeds. Pitch attitude remained relatively constant except at high airspeed (119 KCAS) which resulted in an increasing nose down pitch attitude with increasing descent rates. The control positions at the aft cg configuration (fig. 11), generally followed the same trends as the mid cg configuration.

18. Throughout the control position in trimmed forward tests, control margins in all axes were well in excess of 10%.

STATIC LONGITUDINAL STABILITY

19. Static longitudinal stability characteristics were evaluated with the pitch bias actuator disconnected and FPS off at the conditions listed in tables 1 and 2. These tests were accomplished by trimming the aircraft at the desired airspeed (zero sideslip), then with the collective control fixed, the helicopter was stabilized at incremental airspeeds greater and less than trim airspeeds. Data were recorded at each stabilized airspeed and are presented in figures 12 through 19, appendix F.

20. At all conditions tested, static longitudinal stability was negative to neutral (aft longitudinal cyclic control required to stabilize at increased airspeeds). Flights at an aft cg configuration revealed a strong negative static longitudinal stability at both airspeeds tested. The longitudinal cyclic gradient was most unstable at 60 knots true airspeed (KTAS) $V_T/\sqrt{\theta}$ in both the aft cg configuration (fig. 10) (27 kts/in.) and in the baseline configuration (fig. 13) (33 kts/in.). The static longitudinal stability in climbs at 100 KTAS $V_T/\sqrt{\theta}$ (fig. 14) was slightly negative (75 kts/in.) and during descents at 100 KTAS $V_T/\sqrt{\theta}$ (fig. 14) was essentially neutral. Control forces were not measured; however, qualitatively there were very little force cues noted within $\pm$15 knots indicated airspeed (KIAS) of the trim airspeed. With larger variations in airspeed, longitudinal cyclic control forces were more perceptible requiring forward force for decreased airspeed and rearward force for increased airspeed. In this same airspeed range lateral cyclic control forces increased significantly. The measurement of control forces should be considered for any future simulator validation testes. Directional pedal position remained relatively constant at all but the 60 KTAS $V_T/\sqrt{\theta}$ conditions (figs. 12 and 13) which required slight (less than 1 inch) right pedal for increasing airspeed. With the stabilator fixed, the aircraft pitch attitude was increasingly nose down with increasing airspeed. As airspeed was increased and decreased it was noted that rotor speed correspondingly increased and decreased approximately 1% (2-3 RPM) requiring continuous adjustment to maintain a constant $N_R/\sqrt{\theta}$.

21. Test data presented in reference 4 with the stabilator allowed to move as programmed and the PBA failed indicate positive static stability about the trim airspeeds evaluated. The programmed stabilator successfully creates a positive longitudinal static stability gradient.

STATIC LATERAL-DIRECTIONAL STABILITY

22. Static lateral-directional stability characteristics were evaluated in level, climbing and descending flight at the conditions listed in tables 1 and 2. Tests were conducted by stabilizing the aircraft at the trim condition (zero sideslip) and then, with the collective control fixed, incrementally increasing sideslip angles in both directions. Data were recorded at each stabilized sideslip and are presented in figures 20 through 28, appendix F.

23. Apparent static directional stability, as indicated by the variation of directional control position with sideslip, was positive (increasing left directional control with right sideslip) at all conditions tested. Directional control variation with sideslip was essentially linear and the gradient of this curve was airspeed dependent. The gradient of directional control position with sideslip varied from approximately 18 deg/in. at the 60 KTAS $V_T/\sqrt{\theta}$ test conditions (figs. 20 and 21) to 6.4 deg/in. at the 140 KTAS $V_T/\sqrt{\theta}$ test conditions (fig. 22). The gradient at all 100 KTAS $V_T/\sqrt{\theta}$ test conditions was approximately 10 deg/in. Left and right lateral cg (fig. 23), did not significantly affect the directional control position with sideslip gradient at 100 KTAS $V_T/\sqrt{\theta}$. However, the gradient was greater during climbs at 100 KTAS $V_T/\sqrt{\theta}$ (8.75 deg/in.) than during descents (12.1 deg/in.) (fig. 24). Changes in rotor speed, longitudinal cg, and thrust coefficient had very little effect on the gradient of directional control position with sideslip.

24. Dihedral effect, as indicated by the variation of lateral control position with sideslip was positive (increasing right cyclic control required for increasing right sideslip) at all test conditions. Lateral cyclic control position varied linearly with sideslip at all conditions tested; however, the gradient for right sideslips was generally steeper than for left sideslip, requiring more right lateral cyclic per degree of sideslip. The greatest difference occurred at the 140 KTAS $V_T/\sqrt{\theta}$ conditions (figs. 20 and 21). At the left lateral and right lateral cg conditions (fig. 23), the gradient of the lateral cyclic control variation with sideslip curves were nearly identical but were displaced by approximately 1.5 inches of lateral cyclic.

25. Sideforce characteristics, as indicated by the variation of bank angle with sideslip were positive for right sideslip (increasing right roll attitude with right sideslip), and weak but positive for left sideslip at all conditions tested. At lower airspeeds, 60 KTAS $V_T/\sqrt{\theta}$ (figs. 20 and 21), the sideforce

cues were very weak at small sidelip angles (less than $\pm15°$). At the higher airspeed conditions, 140 KTAS $V_T/\sqrt{\theta}$ (fig. 22), the sideforce cues increased with increasing sideslip and were noticeable at even small sideslip angles. At 100 KTAS $V_T/\sqrt{\theta}$ (figs. 23 through 28), the sideforce characteristics were generally weak but positive at small sideslip angles (less than ±10); however, sideforce cues were more perceptible to the pilot during right sideslip than during left sideslip. Using the aircraft attitude indicator and slip ball as references, the pilot was able to readily discern an out of trim condition for right sideslip at much smaller sideslip angles (less than 10°) than for left sideslips. During climbs and descents at 100 KTAS $V_T/\sqrt{\theta}$ (fig. 24), the sideforce characteristics were slightly stonger during climbs than during descents.

LOW-SPEED FLIGHT CHARACTERISTICS

26. Low speed flight tests were conducted at the conditions shown in tables 1 and 2. Surface winds were 3 knots or less during all ground proximity low speed flight tests. A ground vehicle with a calibrated fifth wheel was used as a pace reference for ground proximity tests; the Elliott LASSIE low airspeed system was used for tests at increased altitude. The static control position data are presented in figures 29 through 32, appendix F. Adequate control margins (greater than 10 percent) remained at all conditions tested. Relative wind azimuths flown were 0°, 90°, 180° and 270°. Minimal pilot compensation was required for low speed flight from hover to approximately 15 knots at each relative wind azimuth (HQRS 3). Considerable pilot compensation was required to control pitch, roll and yaw oscillations between approximately 18 and 25 KTAS (translational lift) (HQRS 5). The four per rev vibration of the the main rotor increased at airspeeds between 18 and 25 KTAS to a significant level (VRS 4), then decreased above 25 KTAS to the levels experienced below 18 KTAS (VRS 3). At greater than 25 KTAS, pilot workload decreased in forward and right sideward flight to a level requiring minimal pilot compensation (HQRS 3). Rearward and left sideward flight required moderate pilot compensation (HQRS 4) above 25 KTAS to control pitch, roll and yaw oscillations. The lateral shuffle noted in previous reports (ref 4) was apparent when recovering from left sideward flight.

STABILATOR SWEEPS

27. Stabilator sweeps were performed at constant airspeed in level flight at the conditions listed in tables 1 and 2. Tests were conducted by fixing the stabilator at various positions

throughout the permissible range for the airspeed being flown. Data were recorded at each stablized point, and are presented in figures 33 through 35.

28. Stabilator position was varied from approximately 35° trailing edge down (TED) to 7.5° trailing edge up (TEU) at a target airspeed of 60 KTAS $V_T/\sqrt{\theta}$ (fig. 33) and from approximately 6.5° TED to 7.5° TEU at 140 KTAS $V_T/\sqrt{\theta}$ (fig. 34). Mid and aft cg flights were conducted at 100 KTAS $V_T/\sqrt{\theta}$ (fig. 35). The stabilator position at 100 KTAS $V_T/\sqrt{\theta}$ was varied from approximately 16° TED to 7° TEU. As the stabilator angle increased TED, additional aft cyclic was required, and the aircraft pitch attitude increased nose down. The total pitch attitude change varied from approximately 6° nose down at 100 KTAS $V_T/\sqrt{\theta}$ and 60 KTAS $V_T/\sqrt{\theta}$ to approximately 8° nose down at 140 KTAS $V_T/\sqrt{\theta}$. These pitch attitudes were readily noticeable to all crew members and become very uncomfortable at higher airspeeds at increased trailing edge down stabilator positions. The aft cg configuration at 100 KTAS $V_T/\sqrt{\theta}$ resulted in less than 3° nose up pitch attitude difference from the mid cg configuration at the same airspeed. Lateral, directional and collective positions remained relatively constant under all conditions tested.

DYNAMIC STABILITY

General

29. Dynamic stability characteristics was evaluated in forward flight at the conditions shown in tables 1 and 3. The gust response was evaluated by stabilizing the aircraft at the required conditions, manually locking the stabilator to a scheduled position and disengaging both SAS systems prior to the control input. The long term response was evaluated with all AFCS components operating except the PBA and FPS.

Gust Response

30. Gust response was evaluated by initiating a control pulse in a given axis and observing the aircraft response. Repesentative time histories of control inputs and aircraft response are shown in figures 36 to 39, apppendix F. Longitudinal cyclic and collective pulses resulted in a divergent pitch response in the direction of the pulse (figs. 36 and 37). Minor roll and yaw divergencies were noted. With the lateral cyclic pulses, the aircraft demonstrated a divergence in roll and pitch (fig. 38). The rapidly diverging pitch attitude and pitch acceleration were the primary factors which caused the pilot to initiate recovery.

The coupled divergent response was most pronounced during pedal pulses which caused the aircraft to yaw in the direction of the input, oscillate in roll and accelerate in pitch (up with left pedal; down with right pedal) (fig. 39). These responses were apparent at all airspeeds and conditions tested. With the SAS disengaged, the aircraft would be difficult to fly in moderate to severe turbulence.

Long Term Response

31. The long term response of the UH-60A was evaluated with SAS on and FPS off using the techniques described in reference 6, appendix A. The aircraft exhibited an aperiodic pitch divergence that was random in direction as shown in figures 40 and 41, appendix F. The minimum time for the pitch divergence to develop to a point where recovery was initiated was 14 seconds. The pilot had sufficient time to recognize the divergence and to make a control input to prevent excessive airspeed or pitch attitude changes.

CONTROLLABILITY

32. Controllability tests were performed at the conditions shown in tables 1 and 3 to evaluate the control power, response, and sensitivity characteristics of the UH-60A with SAS off. Controllability was measured in terms of aircraft attitude change (control power), angular velocities (control response), and angular accelerations (control sensitivity) about an aircraft axis following a step control input of a measured size. Following the input, all controls were held fixed until a recovery was necessary. The magnitude of the inputs were varied by using an adjustable control fixture. Summaries of controllability are shown in figures 42 through 44, with sample control inputs shown in figures 45 through 48. Generally, the aircraft exhibited greater control power, response, and sensitivity in all axes SAS off compared to previous SAS on testing (ref 4, app A) and SAS on testing during this evaluation. Times to maximum accelerations and 63% of maximum rates were constant and consistent with those shown in reference 4.

33. Maximum pitch rates were never attained while making longitudinal step inputs SAS off. The aircraft reaction was a nearly pure pitching motion both at hover and forward airspeeds.

34. Lateral step inputs resulted initially in a roll motion followed shortly by yaw, and finally (2 to 5 seconds after control input) a pitching motion. Recovery was intitated as a result of the pitching motion.

35. Directional step inputs in forward flight resulted in a rapid pitch acceleration with yaw immediately after control input (figs. 44, 47, and 48, app F). The pitch acceleration was so strong that the minimum load factor attained during right directional steps was -0.25 compared to the minimum load factor attained during forward longitudinal steps of 0.0.

36. An unusual yaw reaction was consistently noted during left directional steps. Yaw acceleration, as seen in the slope of the yaw rate trace (fig. 47), has a change of direction from left to right approximately one second after control input. It then goes left again after another second, and continues until recovery is initiated.

SPECIAL RSIS MANEUVERS

General

37. A series of special maneuvers was performed to provide data for an analytical determination of stability derivatives. The manuevers were doublets, roll reversals, sideslip reversals, and system identification maneuvers. Test conditions are shown in tables 1 and 3. The stabilator was manually locked at a scheduled position and both SAS turned off prior to initiating a control input. Roll reversal and sideslip reversal test techniques are discussed in the applicable paragraphs below. System identification maneuvers were performed by using a real-time, visual guide displayed as a wave-form on an oscilloscope screen mounted in front of the copilot and at the engineer station. The controls were blocked at the required magnitude by a control fixture. At the start of a control input sequence, a dot showing the current control position and leaving a trace was superimposed on the wave form and traveled to the right at a selected rate of speed. The trace of the actual control input in magnitude (ordinate) and time (abscissa) remained superimposed on the screen at the end of the maneuver to allow an evaluation of the accuracy of the input. Representative time histories of special RSIS maneuvers are shown in figures 49 through 65, appendix F.

Doublets

38. The response of the UH-60A to a single control doublet input about the pitch, roll and yaw axes and along the vertical axis was evaluated as stated in paragaph 34. Time histories of doublets are shown in figures 49 through 53. Doublet inputs in all axes and at all test conditions resulted in a random three axis divergence with small attitude excursions in roll and yaw and

large pitch attitude excursions. Pedal doublets with the left pedal first resulted in a more pronounced right roll attitude coupled with a pitch attitude reflex nose up then down (fig. 51) The excessive nose up or nose down pitch attitude required recovery.

Roll Reversals

39. The response of the UH-60A to a roll reversal was evaluated by establishing a constant bank angle, level turn, disengaging both SAS, and applying a lateral cyclic input opposite the bank angle. The aircraft response to a roll reversal was a highly coupled pitch with roll divergence (fig. 54). The diverging pitch acceleration and combined pitch and roll attitudes required the pilot to initiate recovery.

Sideslip Reversals

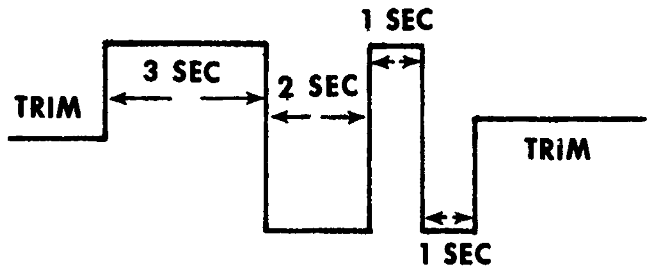
40. The response of the UH-60A to a sideslip reversal was evaluated by establishing a stable level flight condition with a steady heading sideslip. After disengaging both SAS, the sideslip angle was changed from one direction to the other by use of directional pedal while maintaining airspeed with cyclic control. Time histories are shown in figures 55 through 57. The aircraft response to a sideslip reversal was a coupled divergent departure from level flight. The pilot was unable to maintain pitch attitude and airspeed with longitudinal cyclic. The roll acceleration that was perceived in the cockpit resulted in an immediate application of lateral cyclic in the opposite direction of the pedal input. A longitudinal cyclic input was required one-half to two seconds after the pedal application to counter the pitch acceleration. On one right pedal input the collective was inadvertently lowered (5 inches in 2 seconds). The resulting aircraft attitude was 40° nose down, 71° right roll and a 6000 foot per minute rate of descent (fig. 57). The primary reason for initiation of recovery procedures following each input was the pitch attitude and pitch acceleration of the aircraft.

System Identification Maneuvers

41. The response of the UH-60A to a series of programmed sequential control inputs about the pitch, roll and yaw axis and along the vertical axis was evaluated using the oscilloscope and techniques in appendix C. The magnitude of the system identification (SI) manuevers was determined by the results of pulse and doublet inputs. The timing of a nominal SI control input sequence was 3-2-1-1. The control input was held for three counts in one

direction followed by an equal amplitude input in the opposite direction for two counts, etc. An example is shown below.

Figure 1. System Identification (3-2-1-1) Input[1]



[1]Single control input with other controls held fixed.

The SI input sequence was modified to 2-3-1-1 for pedal and longitudinal cyclic inputs due to the excessive rates and attitudes caused by the initial three count input.

42. The aircraft nominal response to an SI input at all test conditions, was a three axis coupled divergence in roll, yaw and pitch. All SI inputs except those begun with up collective, left pedal, or forward cyclic terminated in a rapid pitch down acceleration. Representative time histories of SI inputs in each axis are shown in figures 58 through 65, appendix F. The rapidly diverging pitch attitude and pitch acceleration were the primary factors which caused the pilot to initiate a recovery. Even at realtively small input amplitudes, divergence took place so rapidly that a return to trim control position after completion of the maneuver was virtually impossible. It was found that the shortest time the one-count input could be made was approximately two-thirds of a second. This forced the three-count input to be a minimum of two seconds; too long for a successful completion before recovery became necessary.

MISSION MANEUVERS

43. The visual and motion cues and flight techniques to perform specific mission maneuvers were evaluated by flying a series of flight task segments which included basic flight conditions (hover, cruise, descent, etc.) as well as execution of various discrete maneuvers (acceleration/deceleration, quickstop, etc.). The purpose of this investigation was to develop known performance

related techniques for determination of simulator fidelity. The specific goal was to quantify the piloting technique exhibited in flight and to compare it with that exhibited in the simulator for a given flight test. These flights were conducted with all elements of the AFCS functioning except PBA and FPS. Test conditions are listed in table 1 and 3. Each flight task segment was based on the task description and performance standard given in TC 1-135, Utility Helicopter Aircrew Training Manual (ref 8, app A). Mission maneuvers were flown by four different pilots during two flights and each maneuver was repeated at least once. Flight test segments are listed below in table 4.

Table 4. Flight Task Segments

Takeoff to a hover	Low level flight
Hovering turns	Contour flight
Hovering flight	NOE flight
Normal takeoff	NOE quickstop
Maximum performance takeoff	NOE dash
Climbs and descents	NOE high speed pop-up
Acceleration/deceleration	NOE hard break sideward
Straight and level flight	NOE hard turn
Level turns	Masking and unmasking at a hover
Normal approach to a hover	
Landing from a hover	Terrain flight takeoff

44. The following qualitative remarks are made based on pilot observations and comments during the performance of mission maneuvers.

a. A nose down pitching moment was experienced during takeoff and climbout which was reported previously as a deficiency (ref 4).

b. Vibration characteristics at the pilot's seat were excessive during translation from hover to forward flight and the reverse, ' cents, level turns at angles-of-bank greater than 45 degrees, and the NOE hard break sideward and recovery. Vibrations were primarily 4/rev (17.2 Hz) and were significant to the pilot (VRS 7). The excessive vibrations have been reported previously as a shortcoming (ref 4).

c. The overall maneuverability, responsiveness and agility of the aircraft, especially in the NOE environment, was satisfactory.

CONCLUSIONS

45. The successful completion of the simulator validation flight tests was due, in part, to the close cooperation between USAAEFA and AL. The working relationship between them should be continued and expanded (para 10).

46. The measurement of control forces should be considered for future simulator validation flight tests (para 20).

47. The cathode ray tube display used for the system identification maneuvers greatly increased the accuracy of control inputs. Future versions should include the capability to monitor all controls simultaneously to assure that no inadvertant control motions are made (para 37).

APPENDIX A. REFERENCES

1. Letter, AVRADCOM, DRDAV-DI, 13 October 1980, subject: Rotorcraft Systems Integration Simulator (RSIS) Validation Flight Tests. Test Request No. 79-24.

2. Test Plan, USAAEFA Project No. 79-24, *Validation Flight Test of UH-60A for Rotorcraft Systems Integration Simulator (RSIS)*, December 1980.

3. Technical Manual, TM55-1520-237-10, *Operator's Manual, UH-60A Helicopter*, 21 May 1979, with change 13, dated 11 January 1982.

4. Final Report, USAAEFA Project No. 77-17, *Airworthiness and Flight Characteristics Evaluation UH-60A (Black Hawk) Helicopter*, unpublished.

5. Letter, AVRADCOM, DRDAV-DI, 26 June 1981, subject: Airworthiness Release for UH-60A BLACK HAWK Helicopter S/N 77-22716 for Validation Flight Test of UH-60A for Rotorcraft Systems Integration Simulator (RSIS), Project No. 79-24.

6. Flight Test Manual, Naval Air Test Center, FTM No. 101, *Stability and Control*, 10 June 1968.

7. Final Report, USAAEFA Project No. 74-87, *Flight Evaluation of Non-dimensional Static Longitudinal Stability Test Methods*, July 1979.

8. Training Circular, TC 1-135, Aircrew Training Manual, *Utility Helicopter*, 16 January 1981.

APPENDIX B. AIRCRAFT DESCRIPTION

GENERAL

1. The Sikorsky UH-60A (Black Hawk) is a twin turbine engine, single-main-rotor helicopter capable of transporting 11 combat troops plus a crew of three, cargo, and weapons during day, night, visual, and instrument conditions. A complete description of the aircraft is contained in the operator's manual (ref 3, app A). Major features of the helicopter control system are described below.

FLIGHT CONTROL SYSTEM

General

2. The UH-60A utilizes conventional helicopter cyclic, collective, and directional controls powered by a triply redundant 3050 PSI hydraulic system. The pilot and copilot controls have separate paths to a combining linkage for each control axis. The control inputs from the cockpit controls are transmitted by mechanical linkage to hydraulic servos for power assist and then to the mixing unit The mixing unit combines, sums, and couples the cyclic, collective, and yaw inputs and provides proportional output signals to the main and tail rotor controls. Pilot control is assisted by an Automatic Flight Control System (AFCS) comprised of five basic subsystems: Stabilator, Pitch Bias Actuator (PBA), Stability Augmentation System (SAS), Trim System, and Flight Path Stabilization (FPS).

Automatic Flight Control System

General:

3. The Sikorsky UH-60A AFCS is designed to enhance helicopter stability and handling qualities. The system consists of five major subsystems: the SAS, FPS system, trim system, PBA, and stabilator control system. Electronic control of the systems is provided by commands from a digital SAS/FPS computer and a SAS analog amplifier. The SAS provides three-axis rate damping, pseudo attitude retention, and limited turn coordination. The FPS provides three-axis attitude and airspeed hold and is the primary source of automatic turn coordination. The trim system provides control position hold and control forces versus position gradients. The PBA is designed to provide positive static longitudinal stability and contributes to positive maneuvering stability. The stabilator control system automatically positions the stabilator as a function of flight parameters to tailor aircraft pitch attitude and dynamic response.

Stability Augmentation System:

4. The SAS functions to provide three-axis rate damping and pseudo attitude retention. The SAS is a dual system with one subsystem (SAS-1) controlled by the analog SAS amplifier and one subsystem (SAS-2) controlled by the digital SAS/FPS computer. It is redundant in sensors and command signal path; however, both SAS subsystem command signals drive a single SAS actuator in each axis. During normal operation with both SAS-1 and SAS-2 engaged, each provides one-half of the total system nominal gain and one-half of total system control authority. The control authority of each is electrically limited to ±5 percent of total control travel in pitch, roll, and yaw. SAS inputs to the SAS servo valves are additive to provide a total authority of 10 percent. The sum is limited to ±10 percent authority by mechanical limits of SAS actuator travel. Selectable operation of either SAS-1 or SAS-2 is available at the center console and switching either subsystem OFF automatically doubles the gain of the remaining SAS while its authority remains at 5 percent. All three axes provide rate damping and lagged rate damping (pseudo attitude retention). A washout of the rate damping signal is incorporated in the pitch and yaw channels to prevent saturation during a steady turn.

5. The SAS-1 is controlled by the SAS-1 analog amplifier which continuously derives commands based on inputs from the No. 1 yaw rate gyro, the No. 1 pitch rate gyro, a roll rate signal derived from the No. 2 vertical gyro, and the No. 1 filtered lateral accelerometer signal. The SAS-2 is controlled by the SAS/FPS digital computer. SAS-2 commands are continuously generated in response to signals from the roll rate gyro, No. 2 pitch rate gyro, signals derived from magnetic compass gyros (yaws rate), No. 1 vertical gyro (pitch and roll rate), and No. 1 filtered lateral accelerometers. At airspeeds above 60 knots indicated airspeed (KIAS), input signals from the No. 1 filtered lateral accelerometer and the No. 1 vertical gyro (derived rate) are provided to the SAS-2 system to stabilize yaw during coordinated turns.

6. SAS-2 operation is continuously monitored by the SAS/FPS computer. This monitor system compares inputs from independent sources to SAS command and to SAS actuator output. Failure of any of these comparison checks in SAS-2 input or output indicates a SAS-2 failure (pitch, roll, or yaw channel) and the control input from the affected channel will be removed (actuator remains at failed position) and the SAS-2 advisory light will be illuminated. SAS-1 does not contain fault detection logic.

Flight Path Stabilization System:

7. The FPS is primarily an aircraft attitude hold system that incorporates conditional capability for airspeed hold and turn coordination. The FPS works through the roll, pitch, and yaw trim actuators. The FPS can drive the cockpit control to any position to which the pilot/copilot can turn the controls, resulting in a 100 percent FPS parallel control authority. The AFCS limits the rate of FPS within the maximum override force limits stated in the trim system section. Since FPS inputs drive the cockpit controls through the trim actutors, the TRIM must be ON in order to have FPS.

8. The attitude hold function of the FPS is designed to maintain a desired heading or pitch and roll attitude. The trim attitude, once established, is automatically maintained unless changed by the pilot. At airspeeds greater than 60 KIAS the pitch axis of the FPS seeks to maintain the airspeed for which the trim attitude has been established. When the reference pitch attitude is changed a time delay in the airspeed hold function allows time to stabilize at the new trim airspeed prior to initiating the airspeed hold function. During this time the attitude hold function maintains the pilot-selected pitch attitude.

9. The FPS provides two yaw channel functions: heading hold and automatic turn coordination. For heading hold (below 60 KIAS), the aircraft is maneuvered to the desired heading with the pilot's or copilot's feet depressing one or both of the pedal switches. When the pilot or copilot removes his feet from the switches the aircraft automatically maintains that reference heading. At airspeeds greater than 60 KIAS the coordinated turn feature of the FPS is operational. The coordinated turn feature is initiated by a lateral stick displacement of approximately 1/2 inch and a bank angle of greater than 2 degrees. The feature is disengaged when the bank angle is less than 1 degree and the roll rate is less than 2 degrees per second. Turn coordination is accomplished by directional control inputs through the yaw trim actuator to zero the side force as sensed by the lateral accelerometers in the stabilator control system. At airspeeds greater than 60 KIAS, heading hold is automtically engaged unless the pilot engages the turn coordination feature.

10. The FPS and all inputs are subject to a number of cross-checks, within the computer. In essence, each input (i.e. attitude, rate, airspeed, etc.) is compared either against another independent source of the same information or, in the case of rate inputs, a computer-derived rate. If these comparisons exceed the preprogrammed tolerance, the malfunctioning portion of the

FPS will be disabled and the appropriate AFSC advisory light and the FPS FAIL caution light will be illuminated.

Trim System:

11. The trim system provides zero force control centering at a pilot/copilot selected trim control position, a spring breakout force plus gradient and a pedal damper force. The trim system is selected by activating the push-on push-off switch, marked TRIM, on the AFCS control panel.

12. With the trim system selected OFF there is no control force gradient or control centering in the cyclic control system or directional control system. Directional control movements will be resisted by a pedal damper which generates an opposing pedal force opposite to the proportional rate of pedal movement. This damping force is electrically generated but is continuously engaged without regard to TRIM switch position. With the trim system ON, directional and lateral control forces are developed in the electromechanical trim actuators. These actuators incorporate an electrically controlled rotary spring assembly which allows the pilot to select the zero force control trim position. The designed maximum override force full opposite control position is 80 pounds in directional and 19 pounds in lateral cyclic control. Longitudinal cyclic control forces are developed in an electrohydromechanical pitch trim actuator with a designed maximum override force of 20 pounds.

13. With the trim system selected ON the pilot/copilot may change the cyclic control trim position through two means: a cyclic trim release switch and a cyclic beep trim switch. The cyclic beep trim switch allows the cyclic control trim position to be changed in one direction at a time at a fixed-rate of travel by electrically driving the trim actuator through the rotary spring assembly. The beep trim switch is a four-position "chinese hat" switch mounted on the cyclic stick grip. Activation of the trim release button switch released the force gradient on the longitudinal and lateral cyclic. The position of the cyclic control when the trim release switch is open (released) becomes the new cyclic trim position. At airspeeds below 60 KIAS, when the pedal switches are closed (any pedal switch depressed), the electronically controlled yaw force gradient spring is repositioned by pedal movement resisted only by the pedal rate damper. When the pilot/copilot removes his feet from the pedals which release the pedal switches, the electronically controlled rotary spring reengages, holding the pedals at the new trim position through the pedal breakout plus gradient spring. Above 60 KIAS the pedal switches and the TRIM REL switch together provide yaw trim release.

14. The SAS/FPS computer monitors the trim system by comparing the commanded trim actuator position to the actual position in all three axes. (Trim actuator position may be commanded by the pilot or by the FPS). If this comparison is out of tolerance, the trim system is shut off in the defective axis and the TRIM FAIL caution light and TRIM advisory light on the AFCS computer are illuminated. The trim system may be reset by pressing both POWER ON RESET buttons on the AFCS control panel.

Pitch Bias Actuator:

15. The PBA is an electromechanical differential actuator built into the longitudinal cyclic control system to assure a stable gradient of longitudinal cyclic control position versus airspeed. It receives airspeed, pitch attitude, and pitch rate inputs from the SAS/FPS computer continuously whenever power is applied to the aircraft assuming the SAS/FPS computer detects no faults prejudicial to PBA function. The AFCS control panel switch configuration will not change the PBA function in normal operation. Airspeed signals do not affect the PBA operation below 80 KIAS. PBA inputs do not feed back to the cockpit controls. Since the PBA is, in effect, a variable length control rod which changes the relationship between longitudinal cyclic control and swashplate tilt.

16. The authority of the PBA is 15 percent of longitudinal cyclic full throw and is limited by the computer to a maximum rate of 3 percent per second. PBA function is monitored by the SAS/FPS computer by an actuator feedback system. If actuator position differs from the commanded position by more than the predetermined tolerance, power is removed from the PBA, the actuator remains in the position it was in at the time of failure, and the PITCH BIAS FAIL caution light is illuminated. This could result in loss of up to 15 percent (1.5 inches) of forward or aft cyclic control authority. Intermittent PBA failures due to an actuator position versus command "no compare" may be reset by pushing both POWER ON RESET buttons on the AFCS control panel.

17. The PBA operation may be faded or degraded by "no compare" results in airspeed, pitch rate, vertical gyro inputs, internal mechanical failure, or various SAS/FPS computer failures. A pitch rate or vertical gyro failure results in the PBA centering. An airspeed failure results in a constant 120-knot airspeed signal. A mechanical failure of the PBA causes the actuator to remain in the position in which it failed.

Stabilator Control System:

18. The stabilator control system is an electrically controlled and activated system. The primary purposes of the system are to schedule stabilator incidence to eliminate excessively nose-high attitudes at low airspeed due to downwash impingement on the stabilator, and to optimize pitch attitudes for climb, cruise, and autorotational descent. The control system is composed of two analog amplifiers which operate from independent input sources and command the position of two electric jackscrew actuators acting in series. During normal operation these jackscrews operate in unison, with each providing one-half of the stabilator position input.

19. The stabilator position is programmed between 8 ±2 degrees trailing edge up and 38 ±4 degrees trailing edge down as a function of four variables: airspeed, collective control position, pitch rate, and lateral acceleration. The airspeed input primarily allows the stabilator to align with the main rotor downwash during low-speed flight, thus reducing the stabilator download and eliminating excessively nose-high pitch attitudes at low airspeed. The collective control input reduces coupling of pitch attitude to collective in forward flight. Pitch rate and lateral acceleration inputs are designed to improve the dynamic respose of the airframe. Pitch rate inputs to the stabilator system provide a degree of pitch rate damping to supplement SAS-commanded damping. The lateral accelerometer inputs by providing an indication of both side force and yaw angular acceleration, decouple the pitch response to tail rotor thrust changes resulting from changes in the inflow through the tilted tail rotor with sideslip variation.

20. The stabilator system is independent of the other AFCS subsystems although it shares common inputs. Collective control position airspeed, and lateral acceleration inputs are all dual inputs which are compared in the AFCS computer and the output of the No. 2 pitch rate gyro is compared with a pitch rate derived in the AFCS computer. If the AFCS computer detects a "no compare" in those inputs, the appropriate caution/advisory lights will be illuminated and affected AFCS computer controlled functions will be shut down; however, the AFCS computer effects no control over the stabilator system function.

21. Stabilator malfunctions are detected and controlled within the stabilator amplifier system. The positions of the two actuators are monitored and compared by rate and position. Any system malfunction which causes a minimum difference in actuator position (10 degrees at airspeeds less than 30 KIAS and 4 degrees airspeeds

greater than 150 KIAS) results in an automatic shutdown of power to both actuators. If the malfunction is transient, the stabilator system may be reset by pressing the stabilator AUTO CONTROL RESET button on the AFCS control panel. The pilot may at any time take manual control of the stabilator and control its position by referring to cockpit-mounted stabilator position indicators.

BASIC AIRCRAFT INFORMATION

22. Principal dimensions and general data of the UH-60A helicopter are as follows:

Airframe

Length:

Maximum (rotor blades turning)	64 ft, 10 in.
Fuselage (nose to vertical tail)	50 ft, 0.75 in.
Main rotor to tail rotor clearance	2.8 in.

Width:

Main rotor blades turning	53 ft, 8 in.
Main landing gear	9 ft, 8 in.

Height:

Maximum (tail rotor blades turning)	16 ft, 10 in.
Main rotor clearance (ground to tip, rotor static against stops)	7 ft, 14 in.
Tail rotor clearance (ground to tip, rotor turning)	6 ft, 6 in.

Horizontal Stabilator:

Span	172.6 in.
Chord - at root	44.0 in.
- at tip	30.5 in.
Aspect ratio	4.6
Airfoil section designation root to tip	NACA 0014
Sweep of leading edge, quarter chord	0 deg

Dihedral	0 deg
Range of travel (reference to fuselage water line)	39 deg trailing edge down 38± 4° to 9 deg trailing edge up
Taper ratio	1.87
Area (total)	45.0 sq ft

Vertical Tail:

Span	8 ft, 2 in.
Aspect ratio	1.92
Taper ratio	1.623
Sweep angle (1/4 chord line)	41 deg
Airfoil section designation	NACA 0021 to 65 percent span with 7 deg trailing edge camber lower section
Incidence to fuselage reference line	0 deg
Area (total)	32.3 sq ft

Gross Weight

Maximum alternate gross weight	20,250 pounds
Empty weight	Approximately 10,620 pounds
Primary Mission gross weight	16,260 pounds
Fuel capacity	364 gallons

Main Rotor

Number of blades	4
Diameter	53 ft, 8 in.
Blade chord	1.73/1.75 ft

Blade twist	-18 deg (equiv)
Blade tip sweep	20 deg aft
Blade area (one blade)	46.7 sq ft
Geometric disc area (total)	2262 sq ft
Geometric solidity ratio (blade area/disc area)	0.0826
Airfoil section (root to tip) designation	SC1095/SC1095R8
Thickness (percent chord)	9.5 percent
Main rotor mast tilt (forward)	3 deg
Aspect ratio	15.4
Range of flapping	-6 to 25 deg
Blade droop stop angle (static)	-1/2 deg
(flight)	-6 deg

<u>Tail Rotor</u>

Number of blades	4
Diameter	11 ft
Blade chord	0.81 ft
Blade twist (equiv linear)	-18 deg
Blade area (one blade)	4.46 sq ft
Geometric disc area (total)	95 sq ft
Geometric solidity ratio (blade area/disc area)	0.1875
Airfoil section (root to tip designation	SC1095/SC1095R8
Thickness (percent chord)	9.5 percent
Aspect ratio	6.79
Cant angle	20 deg

Main Rotor RPM

	Power On	Power Off
Minimum	234.7	232.1
Normal	245.0 to 260.5	232.1 to 270.8
Maximum	275.9	283.7
Design	257.9	---

Tail Rotor RPM

	Power On	Power Off
Minimum	1082.7	1070.8
Normal	1130.3 to 1201.7	1070.8 to 1249.3
Maximum	1273.1	1308.8
Design	1189.8	---

Gear Ratios

Main Transmission	Input RPM	Output RPM	Ratio	(Teeth)
Input bevel	29,900.0	5747.5	3.6364	(80/22)
Main bevel	5747.5	1206.3	4.7647	(81/17)
Planetary	1206.3	257.9	4.6774	$\frac{(228 + 62)}{62}$
Tail takeoff	1206.3	4115.5	0.2931	(34/116)
Accessory bevel (generator)	5747.5	11,805.7	0.4868	(37/76)
Accessory spur (hydraulics)	11,805.7	7186.1	1.6429	(92/56)
Intermedite Gearbox	4115.5	3318.9	1.2400	(31/25)
Tail Gearbox	3318.9	1189.8	2.7895	(53/19)

Overall

Engine to main rotor	20,900.0	257.9	81.0419
Engine to tail rotor	20,900.0	1189.8	17.5658
Tail Rotor to main rotor	1189.8	257.9	4.6136

Rotational Speed Signals at 100 Percent

	RPM	Frequency, Hz
Main rotor, N_R	257.89	11,018.6
Power turbine, N_P	20,900	1393.3
Gas producer, N_g	44,700	2135.7

APPENDIX C. INSTRUMENTATION

GENERAL

1. Except for the main rotor blade angle instrumentation, the test instrumentation was installed, calibrated, and maintained by the US Army Aviation Engineering Flight Activity (USAAEFA) personnel. Digital and analog data were obtained from calibrated instrumentation and were recorded on magnetic tape and/or displayed in the cockpit. Recorded data were taken at 94 samples per second, and 30 Hz filters were used on all parametes.

2. The sensitive instrumentation and related special equipment in the cockpit is listed below.

Pilot Panel

Boom airspeed
Boom altitude
Radar altitude
Digital rotor speed
Sideslip
Elliott longitudinal airspeed
Elliott lateral airspeed
Elliott vertical speed
Normal acceleration

Copilot Panel

Stabilator position
Ship airspeed
Ship altitude
Control position scope output

Center Console

Longitudinal cyclic control position
Lateral cyclic control position
Pedal position
Collective control position
Ballast cart controller

Engineer Station

Instrumentation controller
Outside air temperature
Fuel used (Eng 1 & 2)
APU fuel used
Control position master scope and computer
Calculator
Del Norte radio range controller

3. Data parameters recorded onboard the aircraft in PCM format.

Time of day
Pilot's event
Engineer's event
Run number
Main rotor azimuth
Sideslip
Angle of attack
Radar altitude
Boom airspeed
Power turbine speed (Eng 1 & 2)
Gas producer speed (Eng 1 & 2)
Main rotor speed
Fuel flow rate (Eng 1 & 2)
Engine torque (Eng 1 & 2)
Main rotor mast bending moment (2 locations)
Main rotor torque (3 locations)
Tail rotor torque
Longitudinal control position
Lateral control position
Pedal position
Collective control position
Tail rotor pitch
Lateral primary servo
Forward primary servo
Aft primary servo
Lateral mixing unit
Longitudinal mixing unit
Pedal mixing unit
Collective mixing unit
Longitudinal SAS input
Lateral SAS input
Pedal SAS input
Stabilator position
Main rotor blade flapping (4 blades)
Main rotor blade pitch (4 blades)
Main rotor blade feathering (4 blades)
Swashplate position (3 transducers)
Roll attitude
Pitch attitude
Yaw attitude or magnetic heading (as selected)
Roll rate
Pitch rate
Yaw rate
Roll acceleration
Pitch acceleration
Yaw acceleration

Nose linear acceleration (3 orthogonal axes)
CG linear acceleration (3 orthogonal axes)
PBA position
Fuel used (Eng 1 & 2)
APU fuel used
APU fuel temperature
Boom pressure altitude (2 channels)
Fuel temperature (Eng 1 & 2)
Ballast cart position
Outside air temperature
Del Norte radio range (3 channels)
Elliott low airspeed system (7 parameters)

4. Locations of various transducers are as shown below.

Parameter	FS	BL	WL
Elliott Probe	248	+70	265
Nose Accelerometers	178	-10	215
CG Accelerations	389	-31	207.7
Attitude Gyros (ship)	389	-31	210
Rate Gyros	391	+31	214
Angular Acceleration	391	+31	219
Boom Head	99	+29	190

AIRSPEED CALIBRATION

Boom System

5. The test boom airspeed system was calibrated during level flight using a pace aircraft (T-28) with a calibrated system, and also using a trailing bomb. The position error is presented in figure 1. Altitude was corrected assuming the position error was completely from the static source.

Elliott LASSIE Low Airspeed System

6. The Elliott Low Airspeed Sensing and Indicating Equipment (LASSIE), made by Marconi-Elliott Avionics System, Rochester, Kent, England, was used for the measurement of omnidirectional low airspeeds. The unit consists of a swiveling probe mounted in the rotor downwash (photo 1), an onboard air data computer, and indicators. The rotor downwash assures adequate dynamic pressures on the probe regardless of airspeed. At low airspeeds, the air data computer computes the airspeed primarily by using the

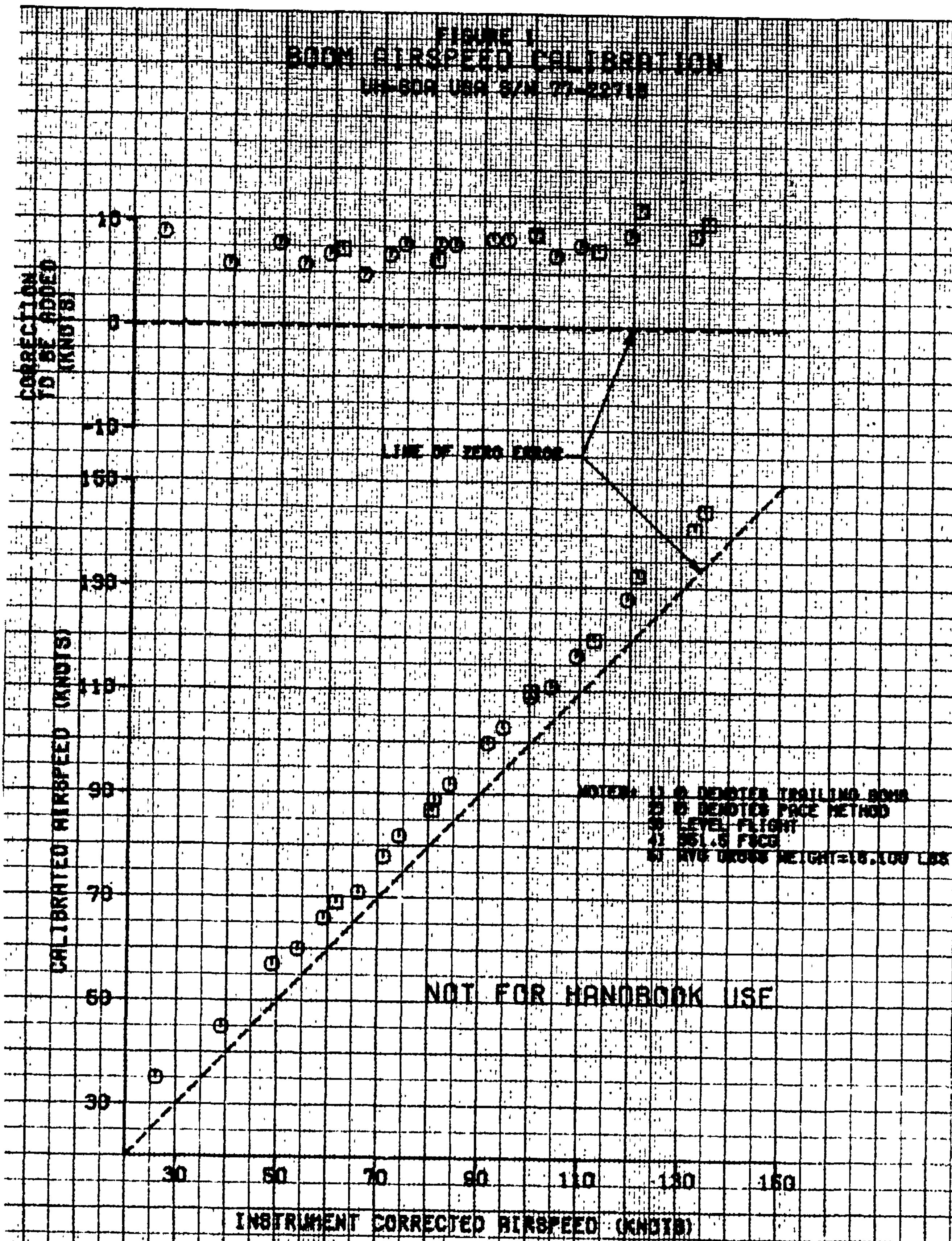


FIGURE 1
BOOM AIRSPEED CALIBRATION
CORRECTION TO BE ADDED (KNOTS)
10
0
-10
LINE OF ZERO ERROR
150
130
110
90
70
50
30
CALIBRATED AIRSPEED (KNOTS)
NOTES: 1) ⊙ DENOTES TRAILING BOMB
2) ⊡ DENOTES PACE METHOD
3) LEVEL FLIGHT
NOT FOR HANDBOOK USE
30
50
70
90
110
130
150
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

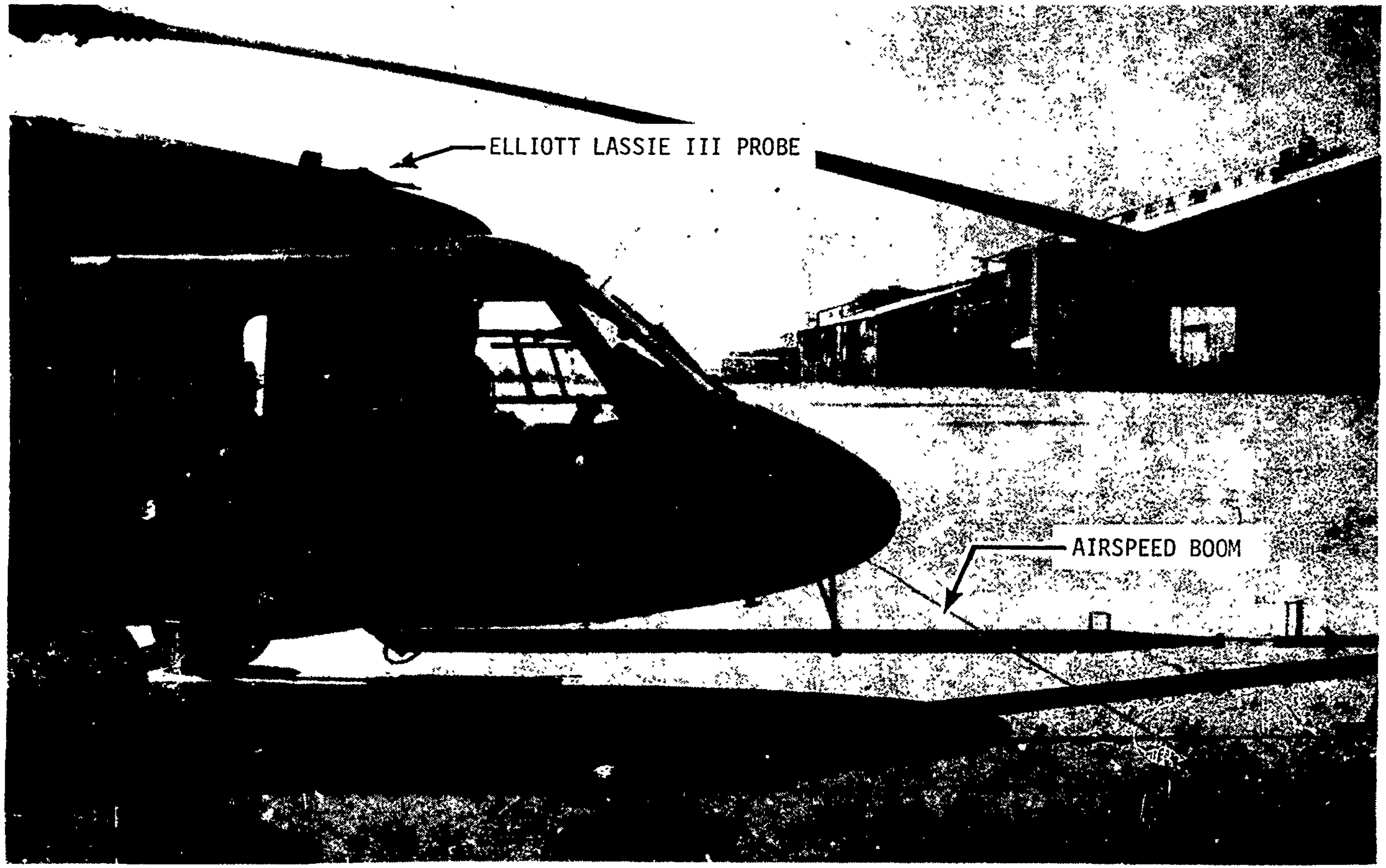


Photo 1. UH-60A Airspeed Measuring Systems

measurement of probe angle. As speed increases, and the probe is relatively less affected by the downwash, the system operates much like a conventional pitot-static system. The transition from low speed to high speed flight (around 35 knots) is highly non-linear, and also shows very high excursions (± 10 knots) on the indicator and on the recorded data. Calibrations for the recorded data and the indicators are shown in figures 2 through 5. A calibrated speedometer attached to a fifth wheel towed behind a pace vehicle was used as a slow speed test reference. Winds were also recorded, and combined with the vehicle velocity to obtain vector components of true airspeed. Those values were then converted to calibrated airspeed.

SPECIAL INSTRUMENTATION

7. The unique requirements of RSIS validation necessitated the use of some uncommon instrumentation, and led to the development of some new instrumentation.

Blade Angle Measurement

8. All three axes of blade motion (pitch, lead-lag, and flapping) were measured on all four rotor blades. The three transducers for each blade were mounted on a fixture leased from Sikorsky (fig. 6). A sample of the output from the blade transducers is shown in figure 7. Because each transducer was not mounted exactly along the axis of blade motion, a transformation was required to resolve measured angles into true angles. Coefficients for the transformation matrix were determined empirically during calibration. Initial calibration was performed by Sikorsky; subsequent calibrations were done by USAAEFA.

Control Position Display

9. The requirement for system identification (SI) control inputs necessitated implementing a real-time visual guide for the pilots to follow during control input. The nominal SI input was a 3-2-1-1 sequence in which a control input was held for 3 counts in one direction, followed by an equal amplitude input in the opposite direction for 2 counts, etc. A system was developed to display a wave form on an oscilliscope for the copilot to use as a guide for input. The wave form guide was displayed on an oscilliscope in both the engineer and copilot stations. The ordinate is scaled in distance of control travel, and the abscissa is scaled in time. At the start of a control sequence, a dot showing the current position of the control is superimposed on the wave form guide, and moves right at a rate of speed determined by the engineer. A

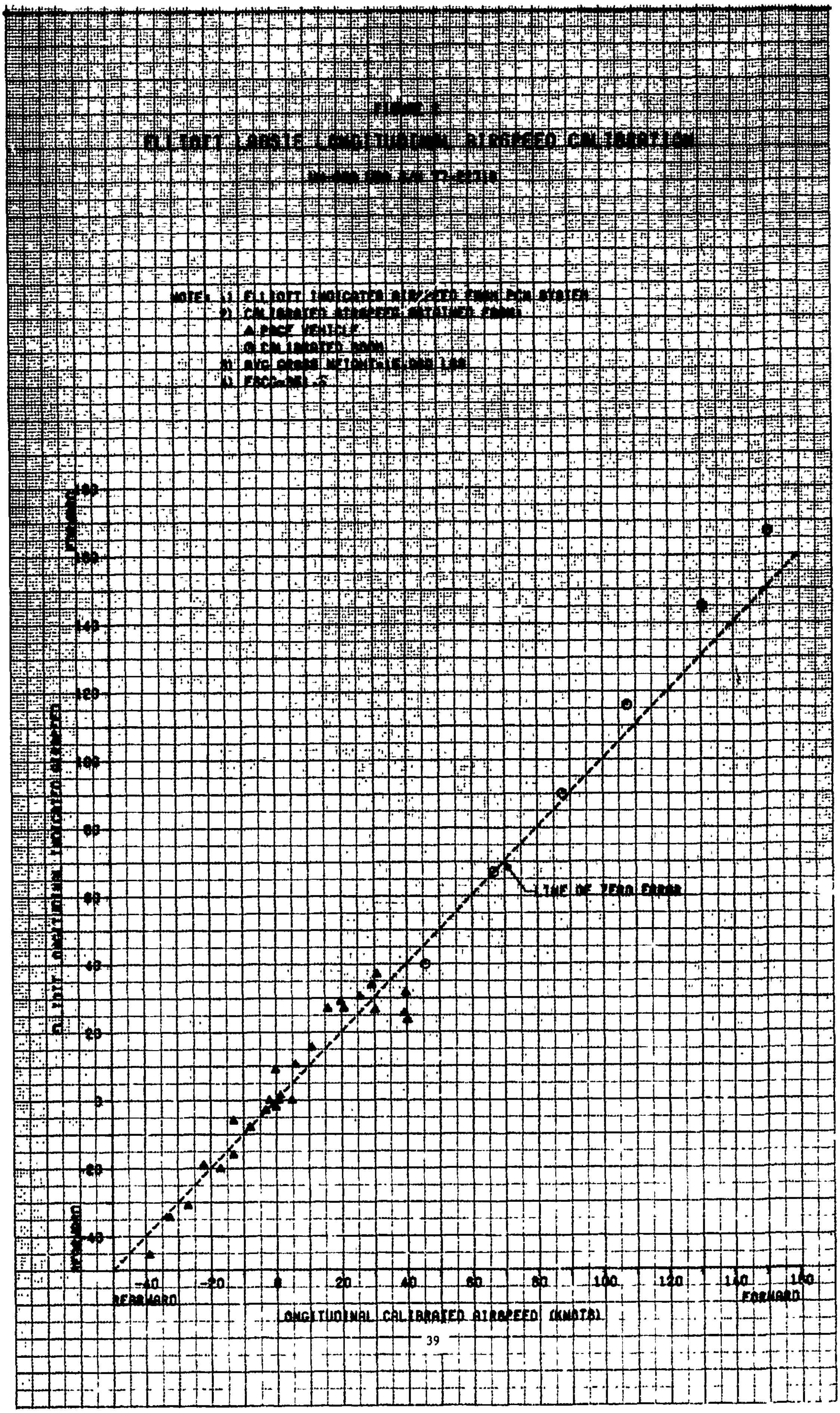

ELLIOTT LADSIE LONGITUDINAL AIRSPEED CALIBRATION
NOTE: 1) ELLIOTT INDICATED AIRSPEED FROM PCM SYSTEM
2) CALIBRATED AIRSPEED OBTAINED FROM:
PACE VEHICLE
CALIBRATED BOOM
LINE OF ZERO ERROR
FORWARD
REARWARD
LONGITUDINAL CALIBRATED AIRSPEED (KNOTS)
-40
-20
0
20
40
60
80
100
120
140
160
120
100
80
60
40
20
0
-20
-40

FIGURE 9

ELLIOTT LASSIE LATERAL AIRSPEED CALIBRATION

UH-60A USA S/N 77-22716

NOTES: 1) ELLIOTT INDICATED AIRSPEED FROM PCM SYSTEM
2) CALIBRATED AIRSPEED OBTAINED FROM PACE VEHICLE
3) AVG GROSS WEIGHT=15,800 LBS
4) FSCG=351.5

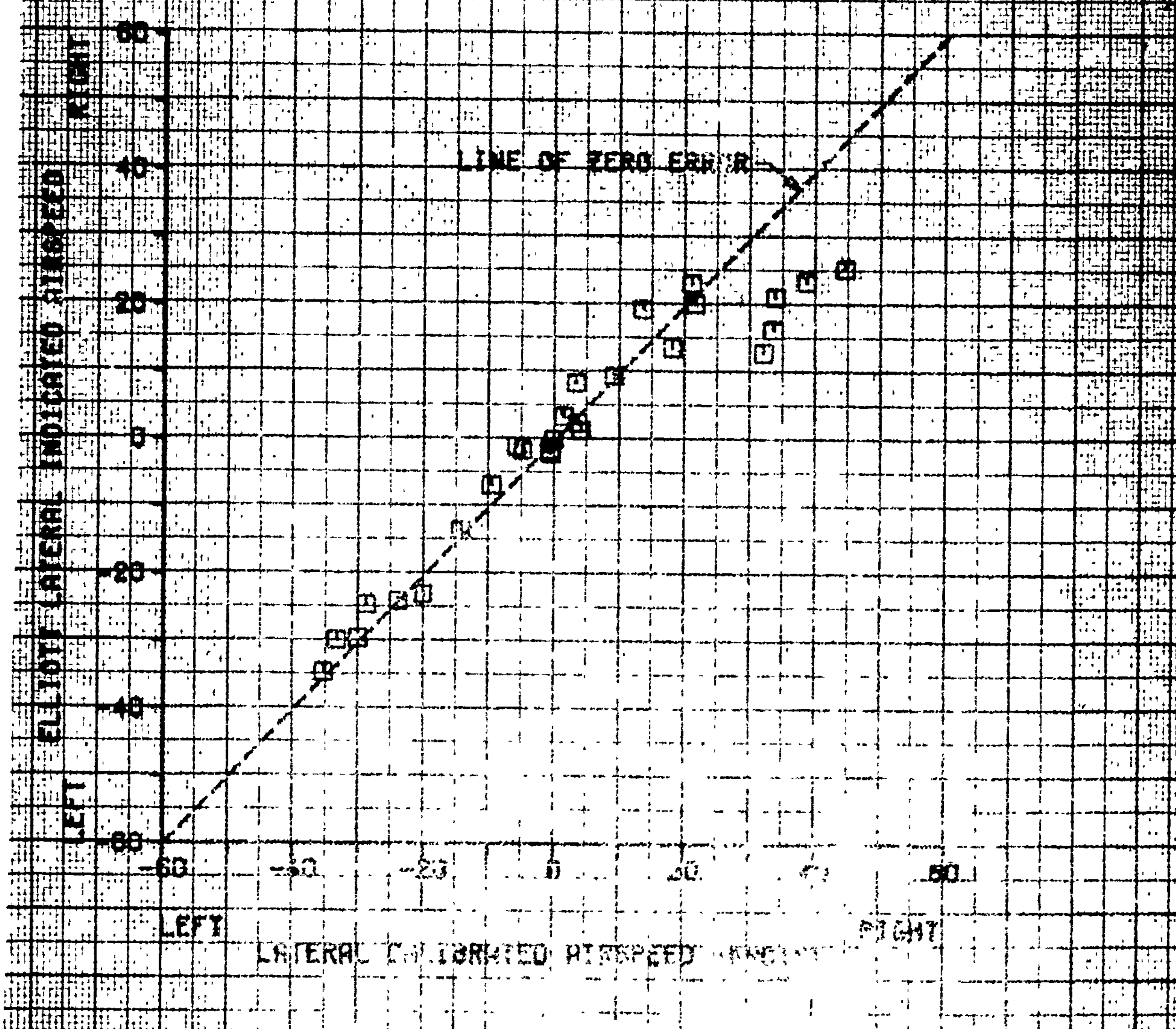


FIGURE 4

ELLIOTT LASSIE LONGITUDINAL AIRSPEED CALIBRATION

COCKPIT INDICATOR

UH-60A USA S/N 77-22716

NOTE: 1) ELLIOTT INDICATED AIRSPEED FROM COCKPIT INDICATOR
2) CALIBRATED AIRSPEED OBTAINED FROM:
▲ PACE VEHICLE
⊙ CALIBRATED BOOM
3) AVG GROSS WEIGHT=16,000 LBS
4) FSCG=361.X

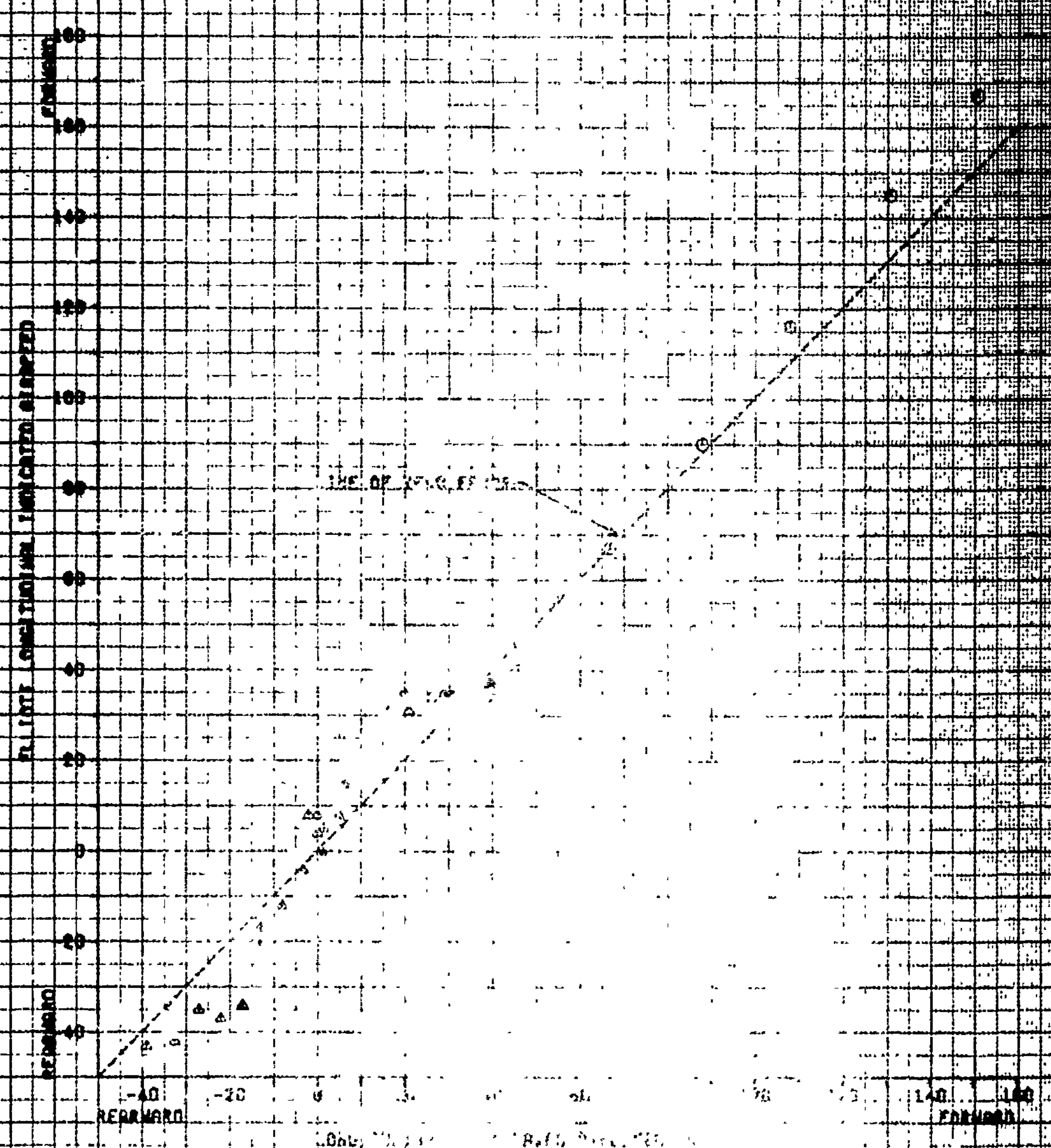


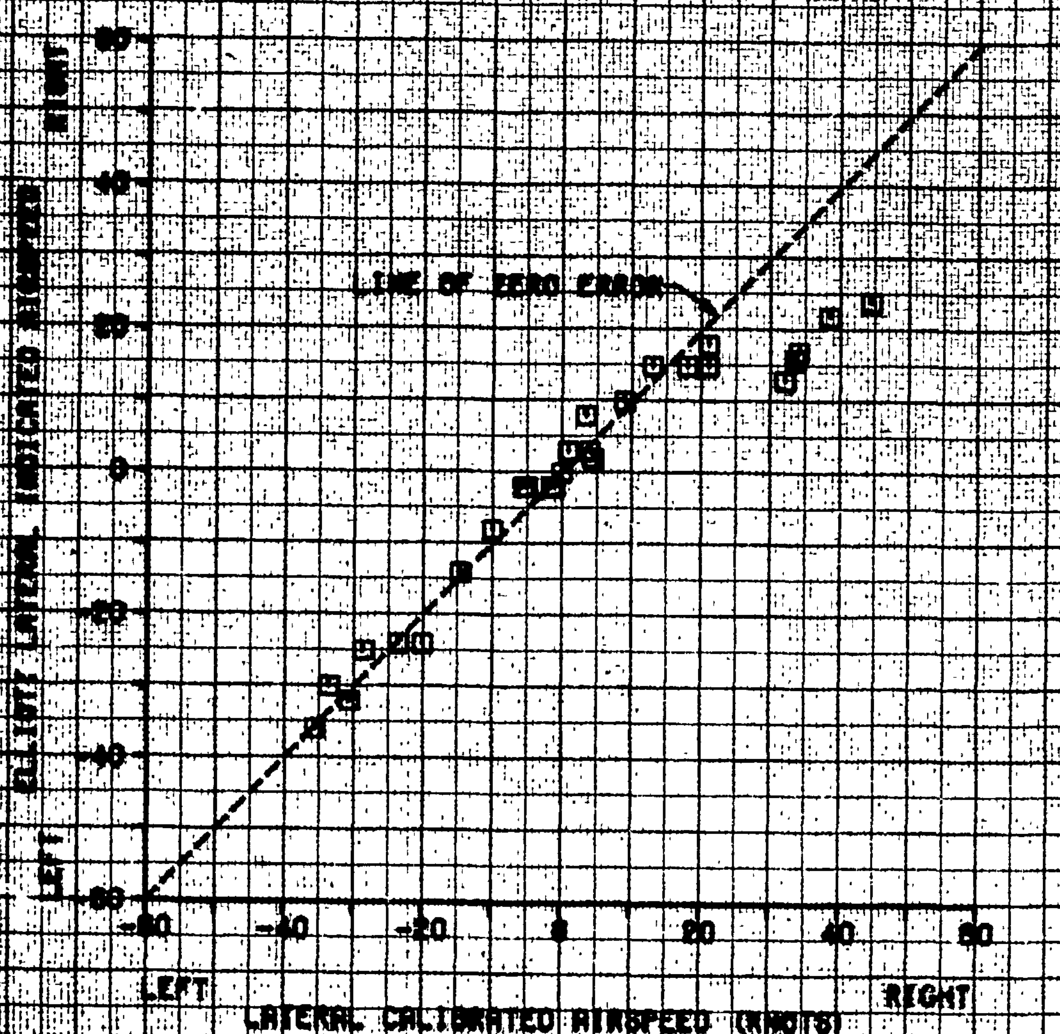
FIGURE 5

ELLIOTT LASSIE LATERAL AIRSPEED CALIBRATION

COCKPIT INDICATOR

UH-60A USA S/N 77-22716

NOTES: 1) ELLIOTT INDICATED AIRSPEED FROM COCKPIT INDICATOR
2) CALIBRATED AIRSPEED OBTAINED FROM PACE VEHICLE
3) AVG GROSS WEIGHT=15,800 LBS
4) FS CG=351.5



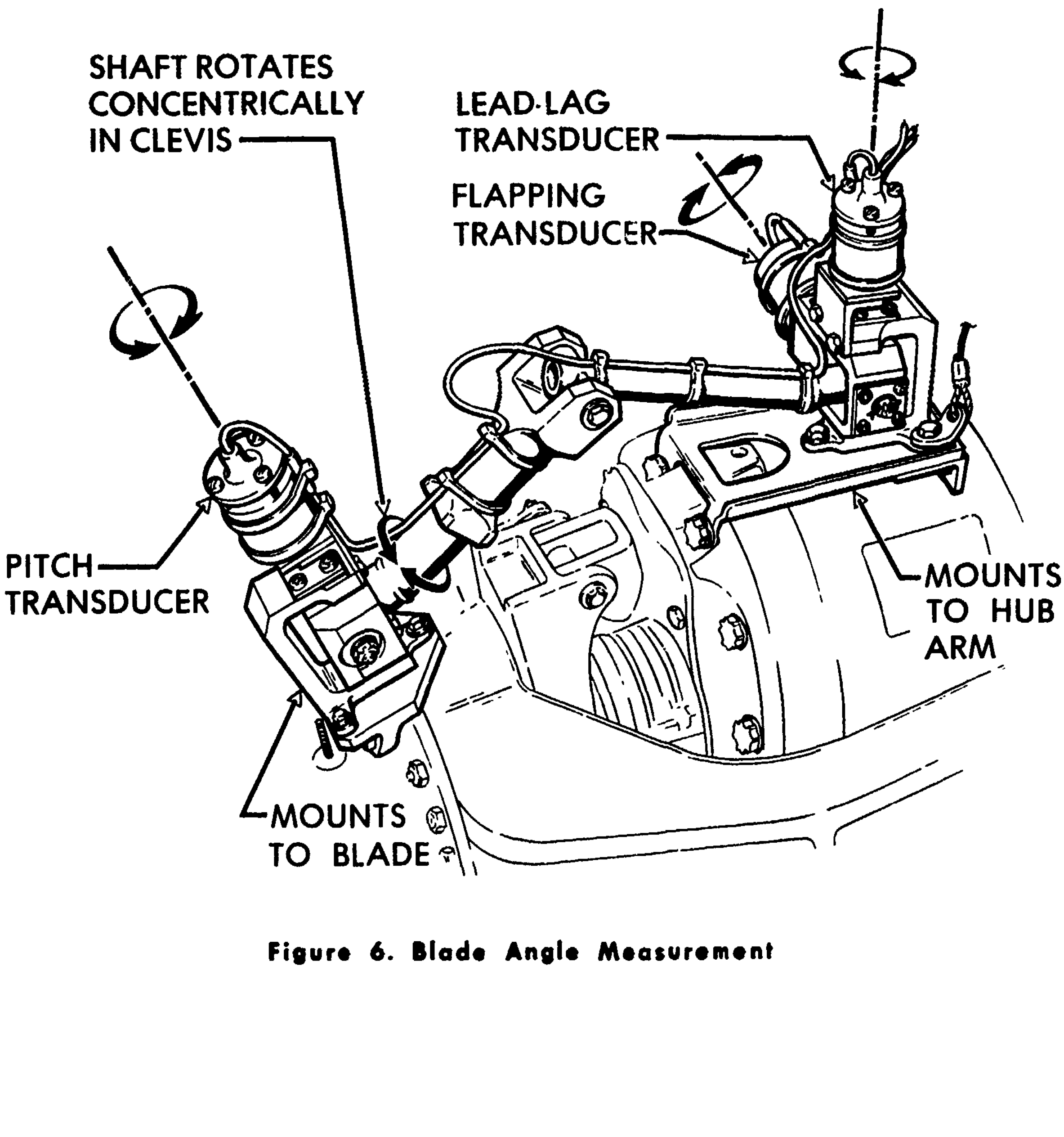
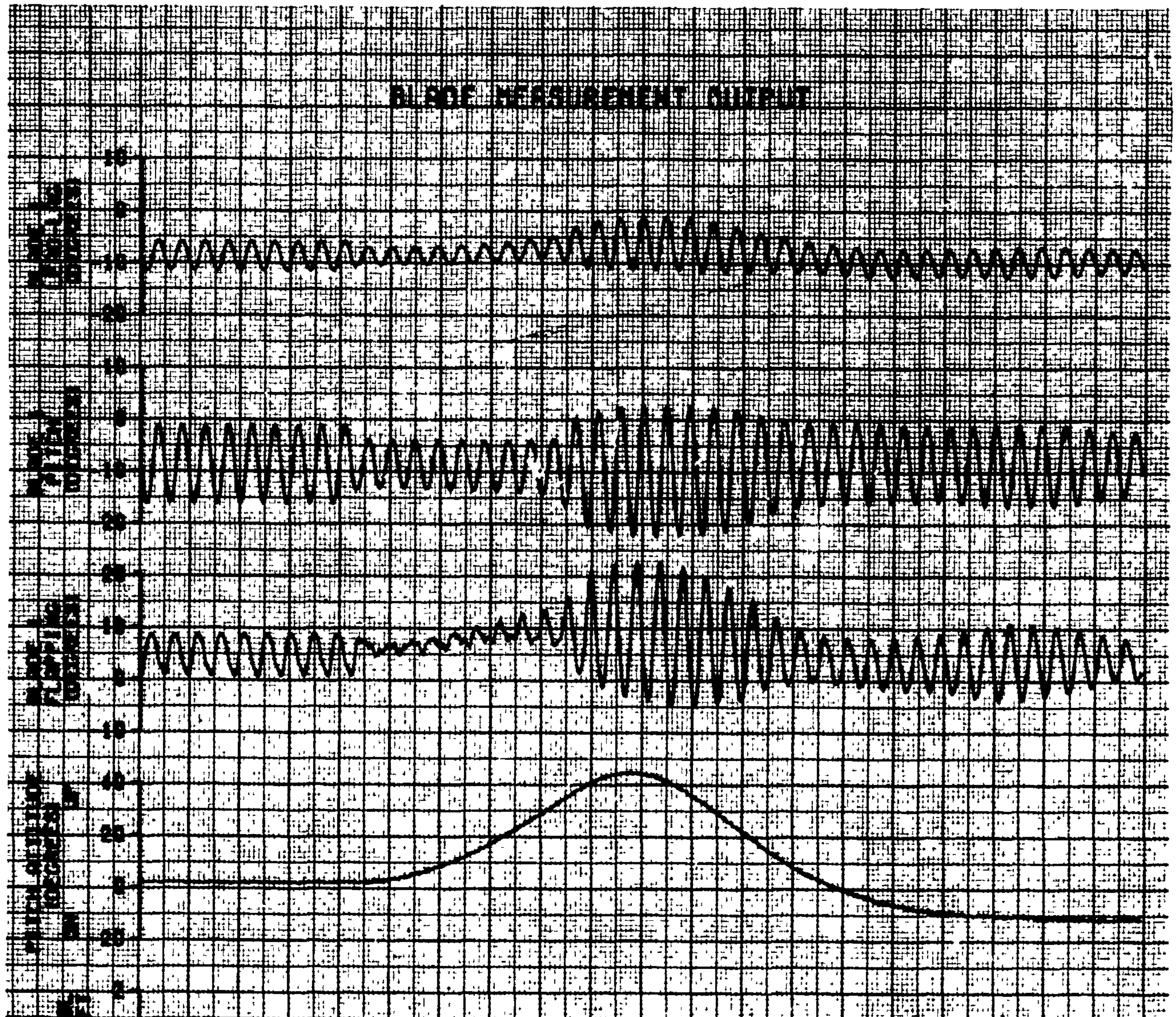


Figure 6. Blade Angle Measurement


BLADE MEASUREMENT OUTPUT
TIME - SECONDS

trace of actual control input remains superimposed on the wave form guide at the end of the maneuver so that judgements may be made as to the adequacy of the input (photo 2).

10. The system consists of an analog-to-digital converter, microcomputer, keyboard, digital-to-analog converter, and two oscilliscopes (photo 3). Eight different wave forms are stored in the nonvolatile read-only-memory (ROM) of the microcomputer. A keyboard assembly is used to select the wave form, its scaling and polarity, and timing. A rotary switch determined which control was displayed.

11. Although the only control input which requires the display is the SI input, it was found that the display was an excellent quality control device for all dynamic maneuvers and static points as well. The real-time display could show control movement during trim points, the crispness and amplitude of steps, and the timing of pulses.

Rotor Azimuth

12. Because blade angle measurements are only meaningful if a correlation can be made of blade location relative to the airframe, a main rotor azimuth measurement was necessary. The rotor azimuth system was designed to provide a continuous stream of parallel binary digital words proportional to instantaneous main rotor shaft position. The circuits do not measure azimuth directly, but process a square wave pulse train whose frequency is proportional to shaft speed, with a one per revolution pulse. Basically, the one/rev pulse resets a counter every revolution, and the pulses from the proportional frequency are counted; each count corresponding to an azimuth position.

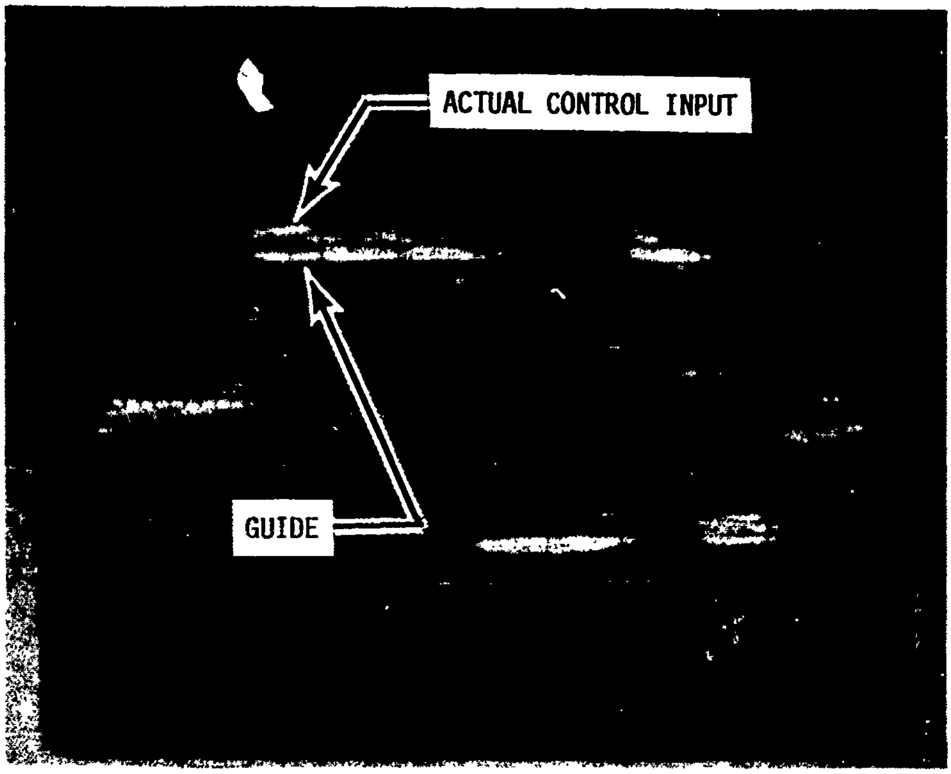


Photo 2. Control Position Display

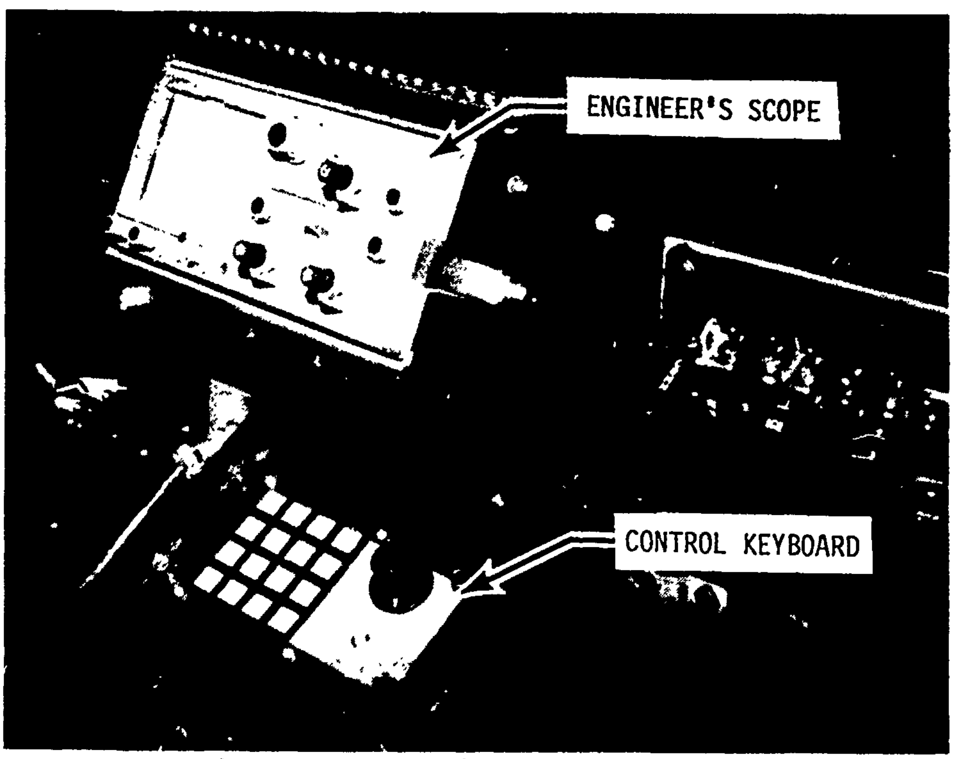


Photo 3. Control Position Display

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. Handling qualities data were obtained using the basic methods contained in the Naval Air Test Center Flight Test Manual FTM 101 (ref 6, app A). Trim points were flown zero sideslip. A Handling Qualities Rating Scale (HQRS) and Vibration Rating Scale (VRS) were used to augment pilot comments (fig. 1 and 2).

AIRCRAFT WEIGHT AND BALANCE

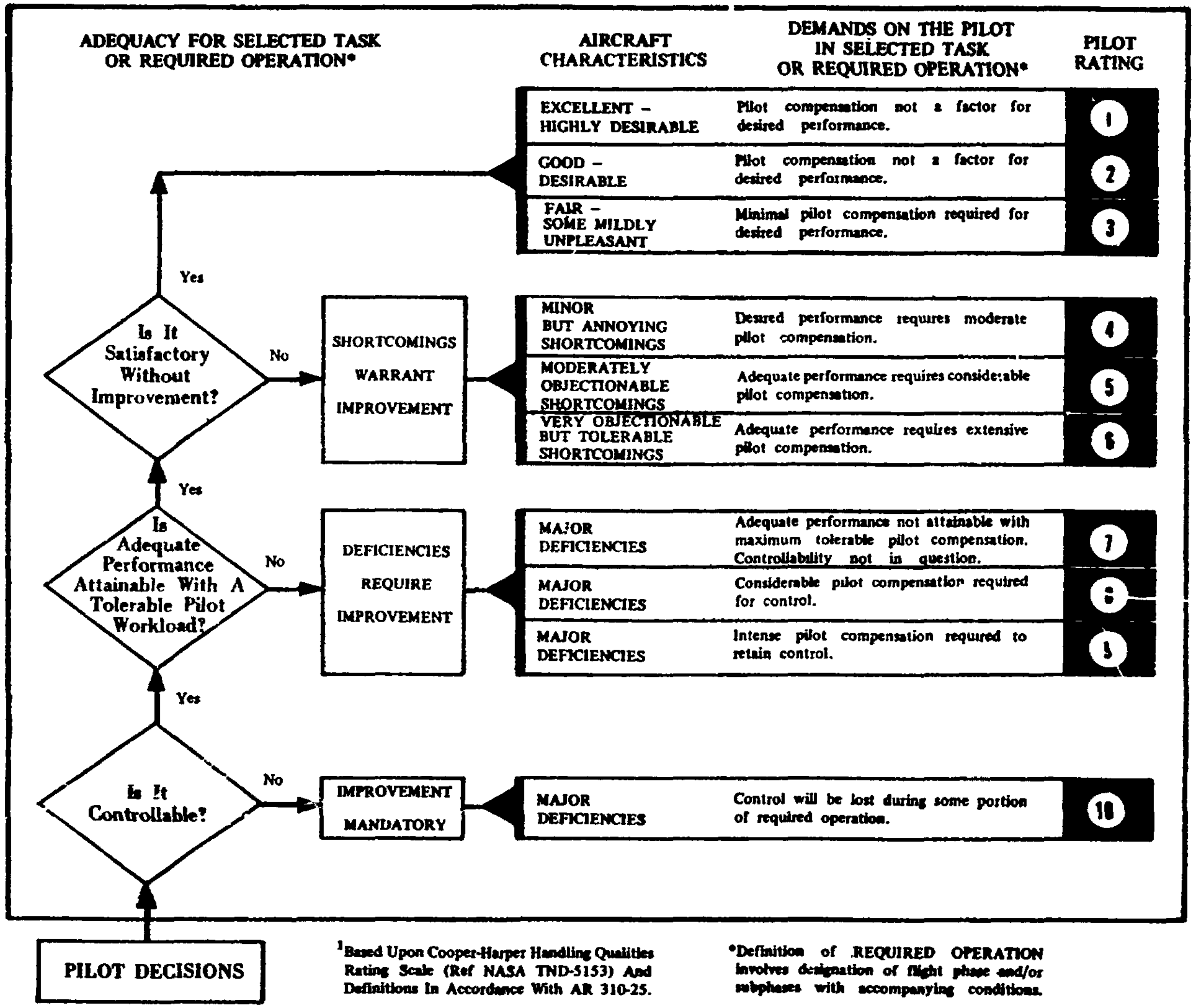
2. The aircraft was weighed in the instrumented configuration with full oil and all fuel drained. The initial weight was 12,579 pounds, with the longitudinal center of gravity located at FS 351.7. The empty moveable ballast cart was located at FS 301. Four empty ballast boxes were installed in the cargo area.

3. Fuel quantity was measured pre and post flight using sight gages calibrated during the Airworthiness and Flight Characteristics program (ref 4, app A). The measured fuel capacity using the gravity fueling method was 364 gallons. The fuel weight for each test was measured prior to engine start and after engine shutdown by using the external sight gage to determine the volume and measuring the specific gravity of the fuel. The calibrated cockpit fuel totalizer indicator was used during the test and was compared with the sight gage readings at the end of each test. Aircraft cg was controlled by a moveable ballast system which was positioned according to a predetermined schedule to maintain a constant cg while fuel was burned. The moveable ballast cart (2600-pound capacity) was attached to the cabin floor by rails and driven by an electric screw jack with a total longitudinal travel of 72.3 inches. Lateral cg was maintained (when necessary) by using the crossfeed fuel control according to a predetermined schedule.

FLIGHT CONDITION

4. The majority of the data collected during this test was taken while maintaining constant aim thrust coefficient (C_T) at specific vehicle and rotor Mach numbers. Aim C_T was maintained using the constant referred gross weight (W/δ), constant referred rotor speed ($N_R/\sqrt{\theta}$) method. Thus, altitude was increased as fuel was burned, and main rotor speed decreased as temperature decreased. Referred true airspeed ($V_T/\sqrt{\theta}$) was also maintained.

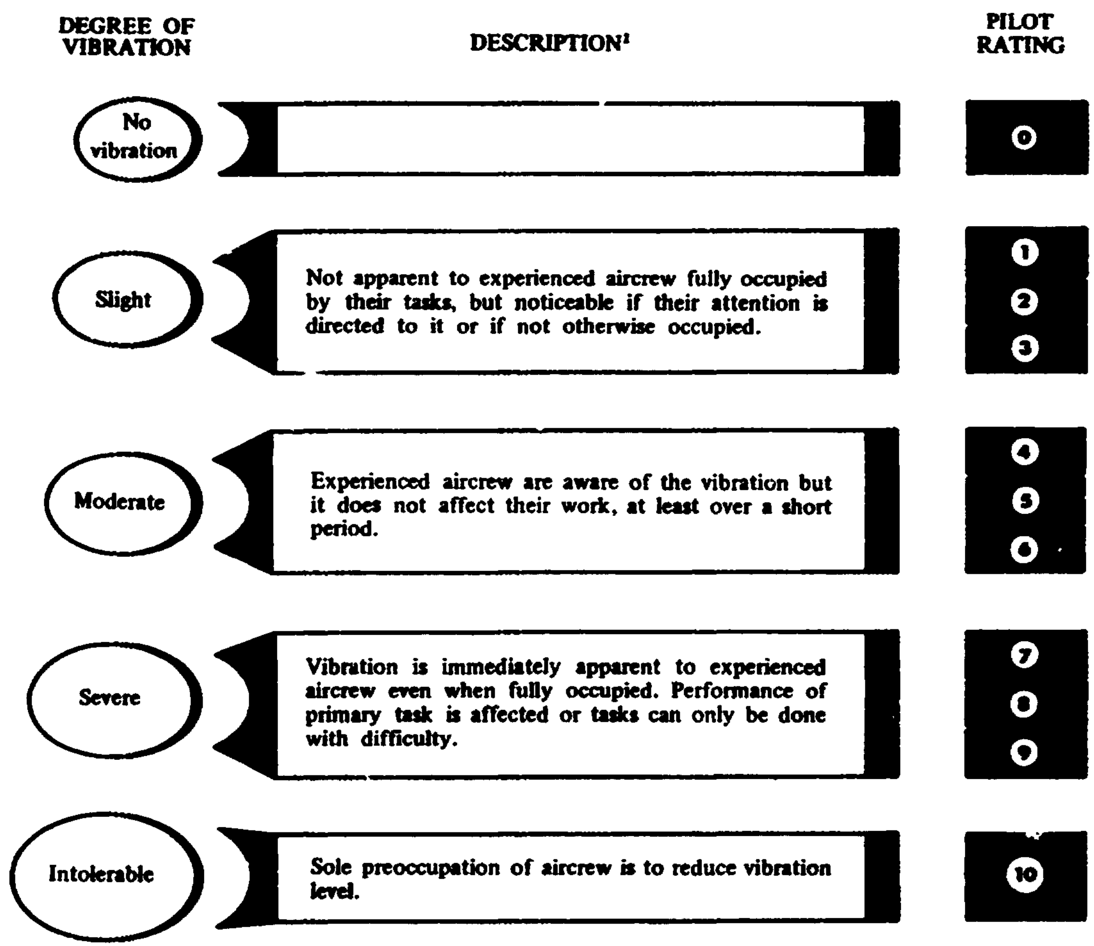
$$C_T = (W/\delta)/\{\rho_o A\ (N_R/\sqrt{\theta})^2(2\pi R/60)^2\} = 0.023556\ (W/\delta)/(N_R/\sqrt{\theta})^2$$



[1]Based Upon Cooper-Harper Handling Qualities Rating Scale (Ref NASA TND-5153) And Definitions In Accordance With AR 310-25.

*Definition of REQUIRED OPERATION involves designation of flight phase and/or subphases with accompanying conditions.

Figure 1. Handling Qualities Rating Scale



[1] Based upon the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Figure 2. Vibration Rating Scale

$$V_T/\sqrt{\theta} = V_{cal}/\sqrt{\sigma}\sqrt{\theta} = V_{cal}/\sqrt{\delta}$$

where:

W = gross weight (pounds)

δ = ambient pressure ratio (ambient pressure/760mm Hg)

ρ_o = standard air density (.0023769 slugs/ft^3)

A = main rotor disk area (2262 ft^2)

N_R = main rotor speed (RPM)

θ = ambient temperature ratio (ambient temperature/288.15°K)

R = main rotor radius (26.833 ft)

$2\pi R/60$ = conversion factor (ft/sec/RPM)

σ = ambient density ratio (ambient air density/.0023769 slugs/ft^3)

5. A programmable calculator (Hewlett-Packard HP-97) was mounted at the engineer's station to provide aim flight conditions during the flight. Values stored at the beginning of a flight were:

a) aim C_T (x 10^4)

b) aim $N_R/\sqrt{\theta}$ (%)

c) engine start gross weight (pounds)

d) fuel density (pounds/gallons)

A sequential input of fuel used in gallons, ambient temperature in degrees Celsius, and $V_T/\sqrt{\theta}$ in knots resulted in an output of pressure altitude in feet, actual rotor speed in percent, and calibrated airspeed in knots. The calibrated airspeed was converted to indicated airspeed using the calibration shown on figure 1, appendix C.

6. Several flights were performed to show the effect of rotor speed on the handling qualities characteristics of the UH-60A. During those flights, the same W/δ was flown as when $N/\sqrt{\theta}$ was 100%. This resulted, of course, in different values for C_T.

TEST TECHNIQUE

7. Except for one ball-centered (coordinated) flight, all trim points were set up at zero sideslip. The programmed stabilator was disabled, and manually slewed to a predetermined setting. During dynamic test maneuvers, trim was estabilished with one of the two redundant SASs turned off. One second before control input, the remaining SAS was disabled. This procedure was necessary because of the inherent instability of the UH-60A with both SASs off; SAS off trims could not be maintained without compensating control inputs. The required control inputs were made by the copilot using a fixture. Steps, pulses, and doublets were made without using the control position display (app C) as a guide. However, the display was used after the manuever to determine adequacy of the control input. For SI inputs, the display was used as both a guide and a quality control aid.

APPENDIX E. PROGRAM MANAGEMENT

1. The Aeromechanics Laboratory (AL) of the US Army Aviation Research and Technology Laboratories contracted with the US Army Aviation Engineering Flight Activity (USAAEFA) to perform validation flight tests on a UH-60A for a research simulator. The test plan was written by USAAEFA from requirements provided by AL. The successful completion of the test program resulted from the cooperation between the two organizations. The close working relationship between USAAEFA and AL should be continued and expanded. In addition to performing the flight test, USAAEFA reduced the data to engineering units on magnetic tape, which were supplied to AL along with data listings and time history plots.

2. The instrumentation list was established jointly by USAAEFA and AL using the data requirements of AL. A shortcoming of the vertical motion simulator (VMS) models located at NASA/Ames is that they do not provide force cues. Control forces were not included on the AL list of requirements and were therefore not measured on the UH-60A. Even though current math models do not provide force cues, future flight tests for model validation should include force measurements so that the models may be upgraded to include forces at a later date.

3. The control position display developed as a timing guide for the SI inputs greatly increased the quality of the inputs. The scope display pictured one control at a time. A possible addition to the capability of the device would be the storage of control position traces for the three controls not used during a maneuver. These traces could then be recalled at the end of a maneuver to assure that no inadvertent "off-axis" control input was made.

4. The flight crew consisted of two experimental test pilots and one flight test engineer. The engineer had the responsibility to operate the instrumentation, calculate flight conditions, modify test card, and program the control position display. One pilot was assigned to fly the aircraft, and the other made control inputs using the fixtures.

APPENDIX F. TEST DATA

INDEX

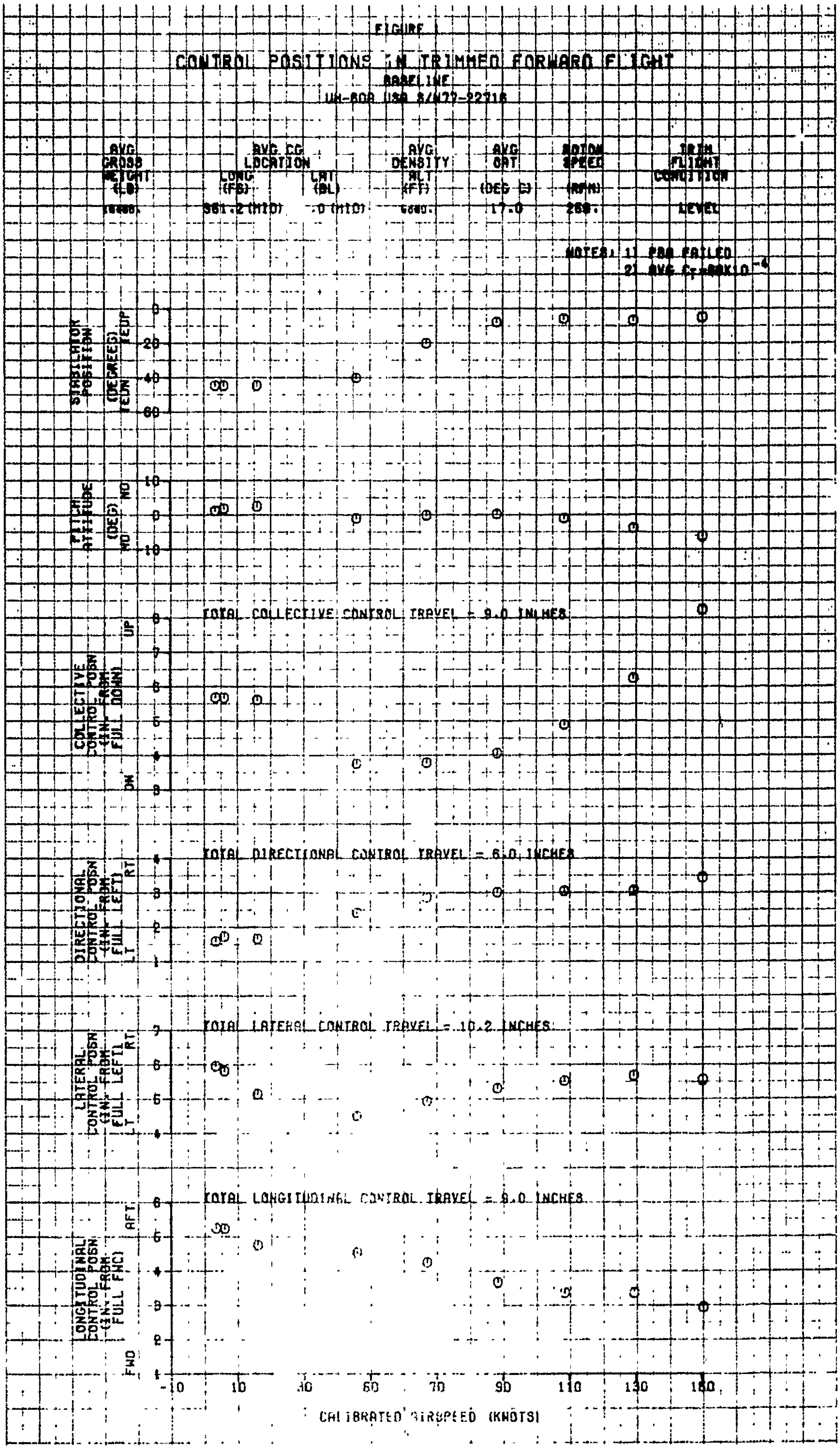
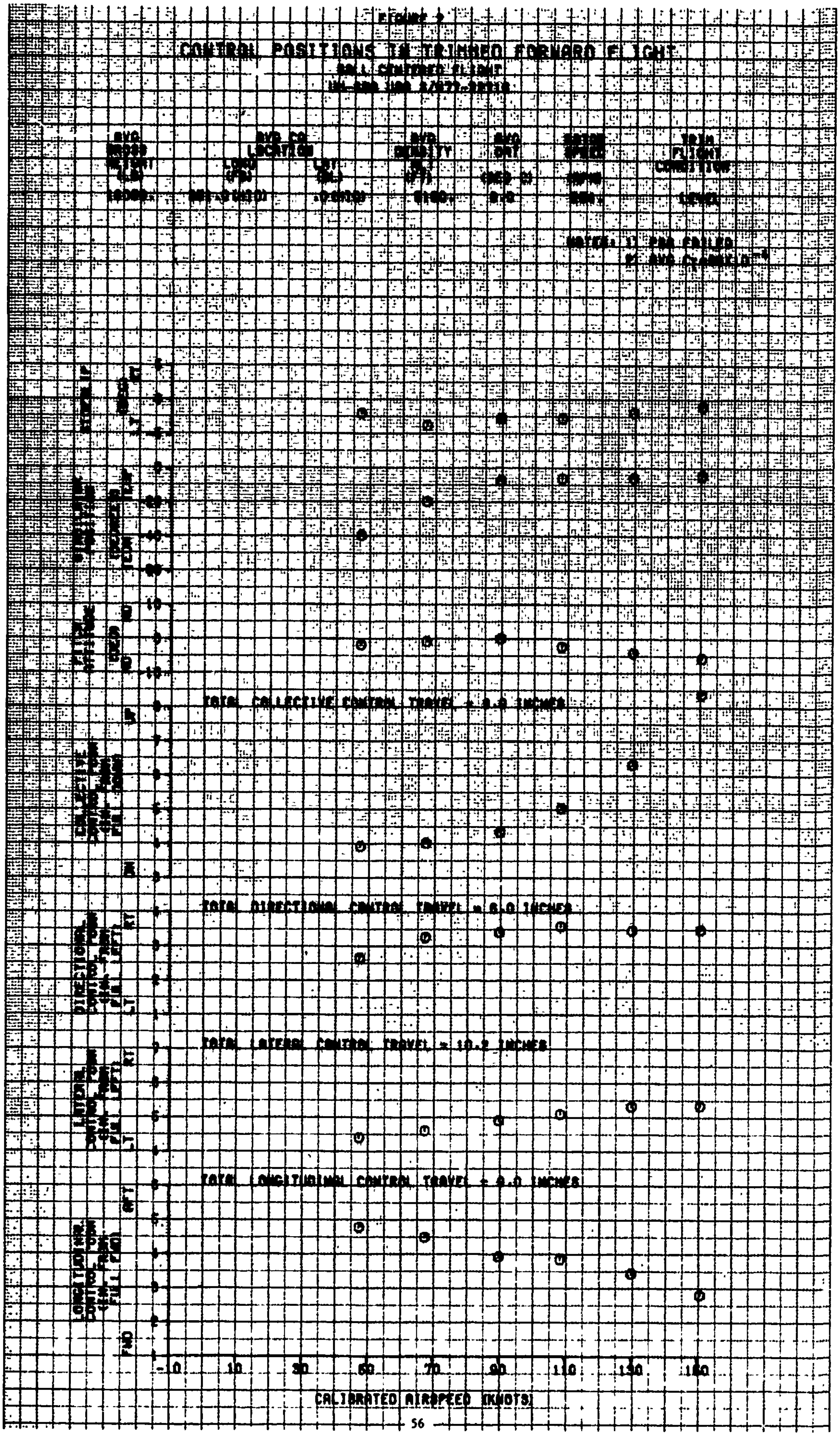


FIGURE
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
BASELINE
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALT (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
TRIM FLIGHT CONDITION
17.0
LEVEL
NOTES: 1) PBA FAILED
STABILATOR POSITION (DEGREES)
PITCH ATTITUDE (DEG)
TOTAL COLLECTIVE CONTROL TRAVEL = 9.0 INCHES
COLLECTIVE CONTROL POSN (IN. FROM FULL DOWN)
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 INCHES
DIRECTIONAL CONTROL POSN (IN. FROM FULL LEFT)
TOTAL LATERAL CONTROL TRAVEL = 10.2 INCHES
LATERAL CONTROL POSN (IN. FROM FULL LEFT)
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.0 INCHES
LONGITUDINAL CONTROL POSN (IN. FROM FULL FWD)
CALIBRATED AIRSPEED (KNOTS)


CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
CALIBRATED AIRSPEED (KNOTS)

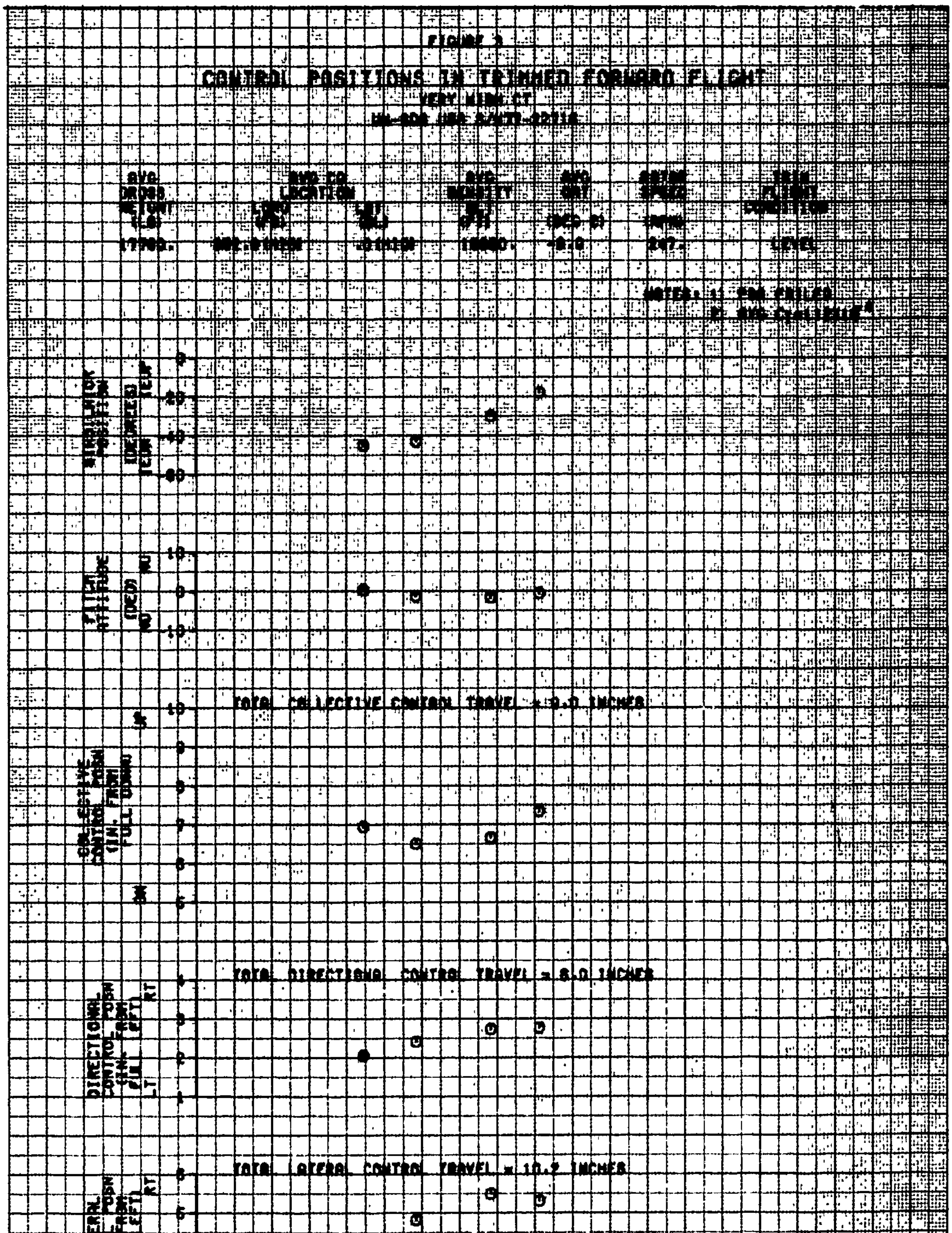
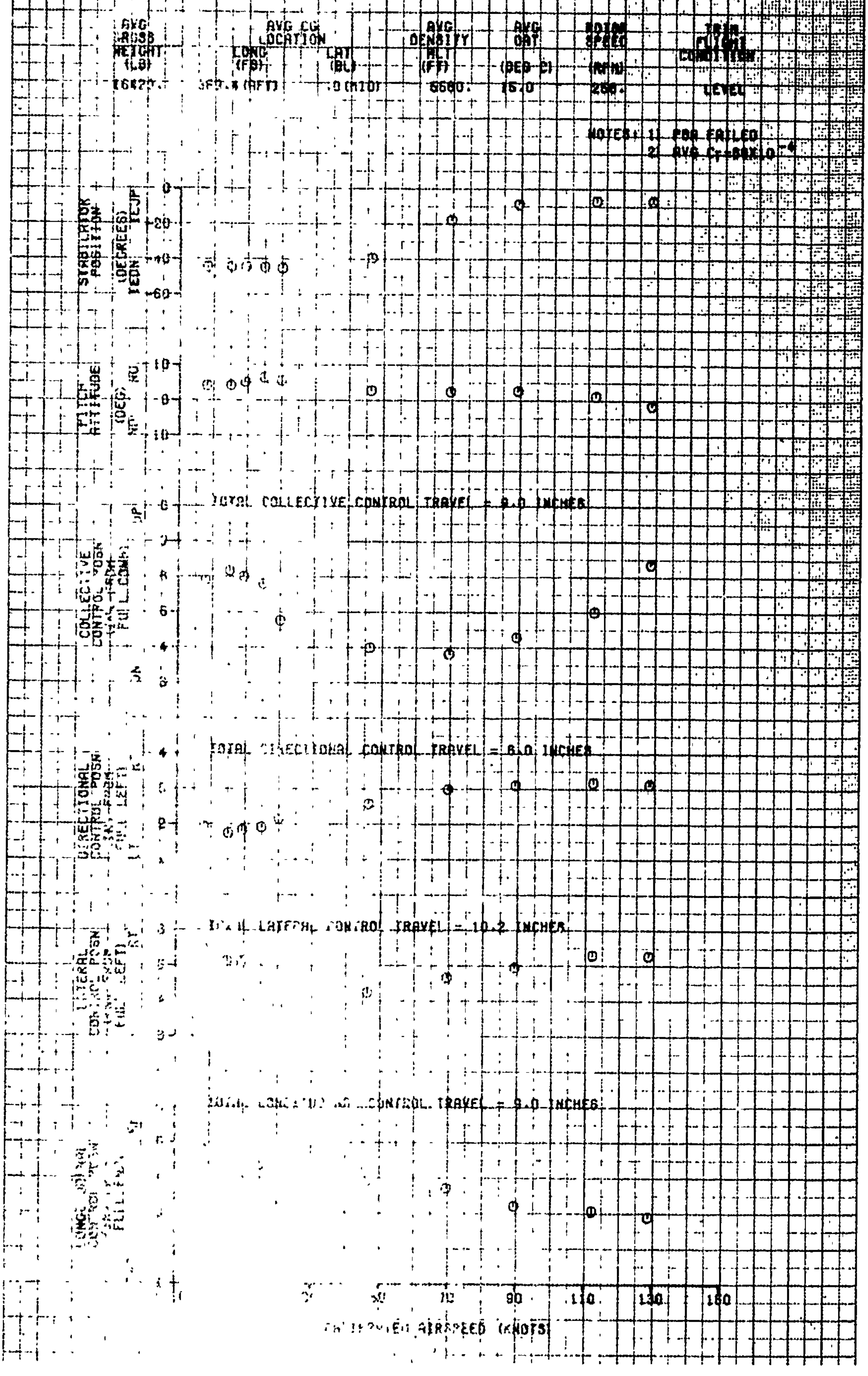

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
AVG. GROSS WEIGHT (LB)
17700.
TRIM FLIGHT CONDITION
LEVEL
STABILATOR POSITION (DEGREES) TEUP TEDN
PITCH ATTITUDE (DEG) NU ND
TOTAL COLLECTIVE CONTROL TRAVEL
COLLECTIVE CONTROL POSN (IN. FROM FULL DOWN) UP DN
TOTAL DIRECTIONAL CONTROL TRAVEL
DIRECTIONAL CONTROL POSN (IN. FROM FULL LEFT) RT LT
TOTAL LATERAL CONTROL TRAVEL = 10.2 INCHES
LATERAL CONTROL POSN (IN. FROM FULL LEFT) RT LT
TOTAL LONGITUDINAL CONTROL TRAVEL
LONGITUDINAL CONTROL POSN (IN. FROM FULL FWD) AFT FWD
-10
10
30
50
70
90
110
130
150
CALIBRATED AIRSPEED (KNOTS)

FIGURE 4

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

AFT CG

UH-60A USA S/N77-22718



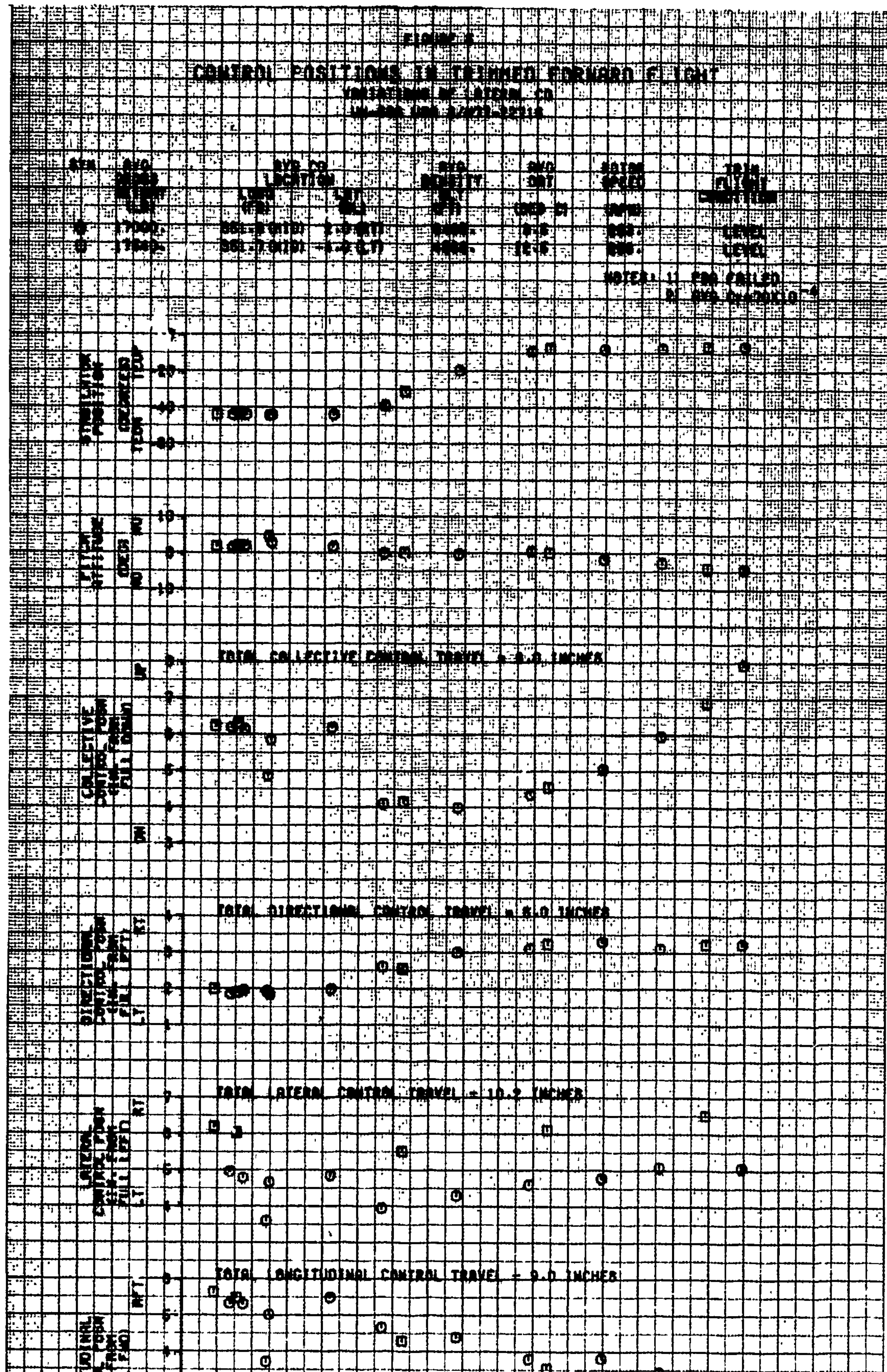

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
TOTAL COLLECTIVE CONTROL TRAVEL = 9.0 INCHES
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 INCHES
TOTAL LATERAL CONTROL TRAVEL = 10.2 INCHES
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.0 INCHES
CALIBRATED AIRSPEED (KNOTS)

FIGURE 8

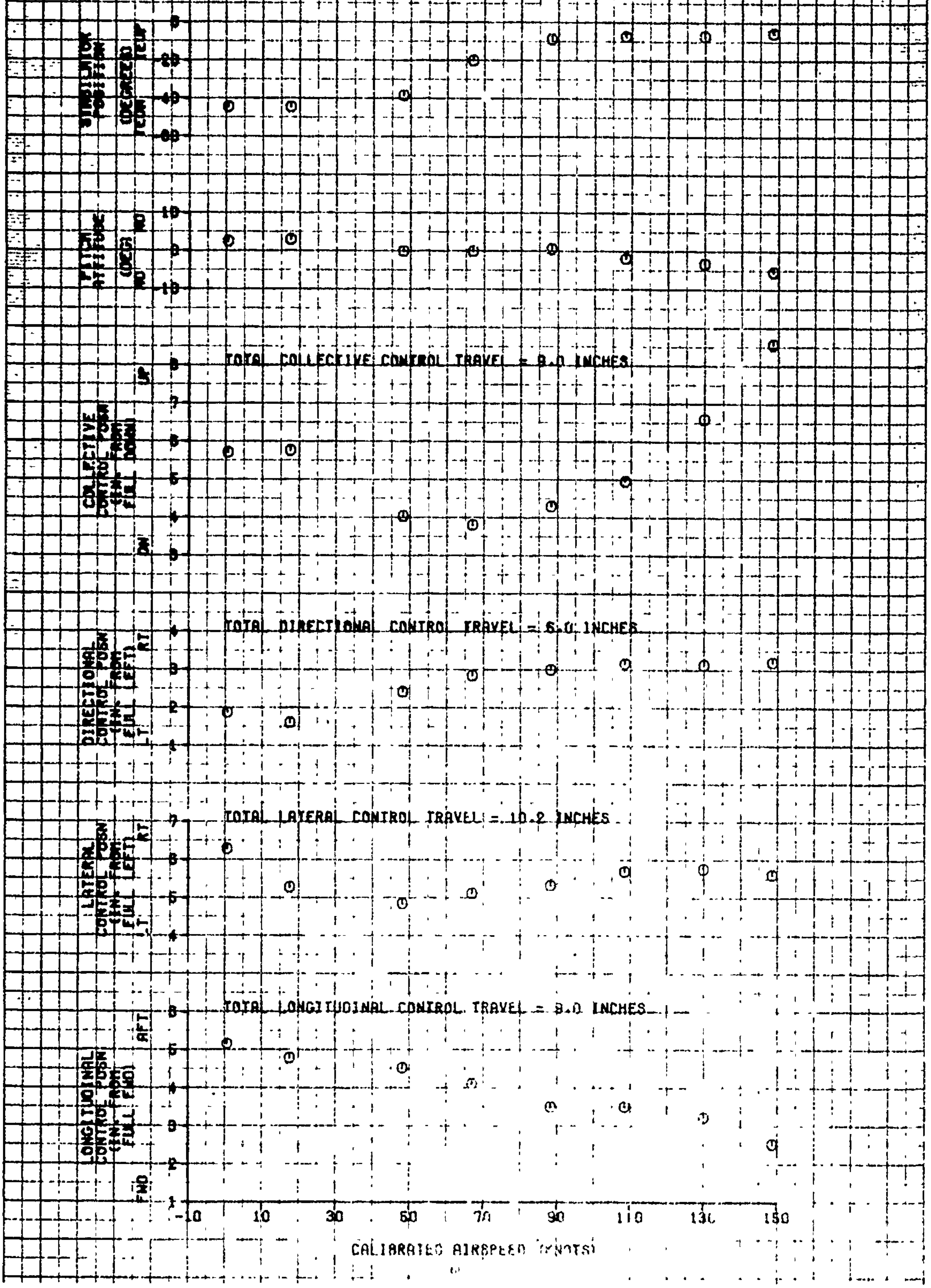
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

96% REFERRED ROTOR SPEED

UH-60A USA S/N77-22716

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
15000.	361.1 (MID)	-.2 (MID)	5040.	11.5	248.	LEVEL

NOTES: 1. PBA FAILED
2. AVG C_T = 80X10^{-4}



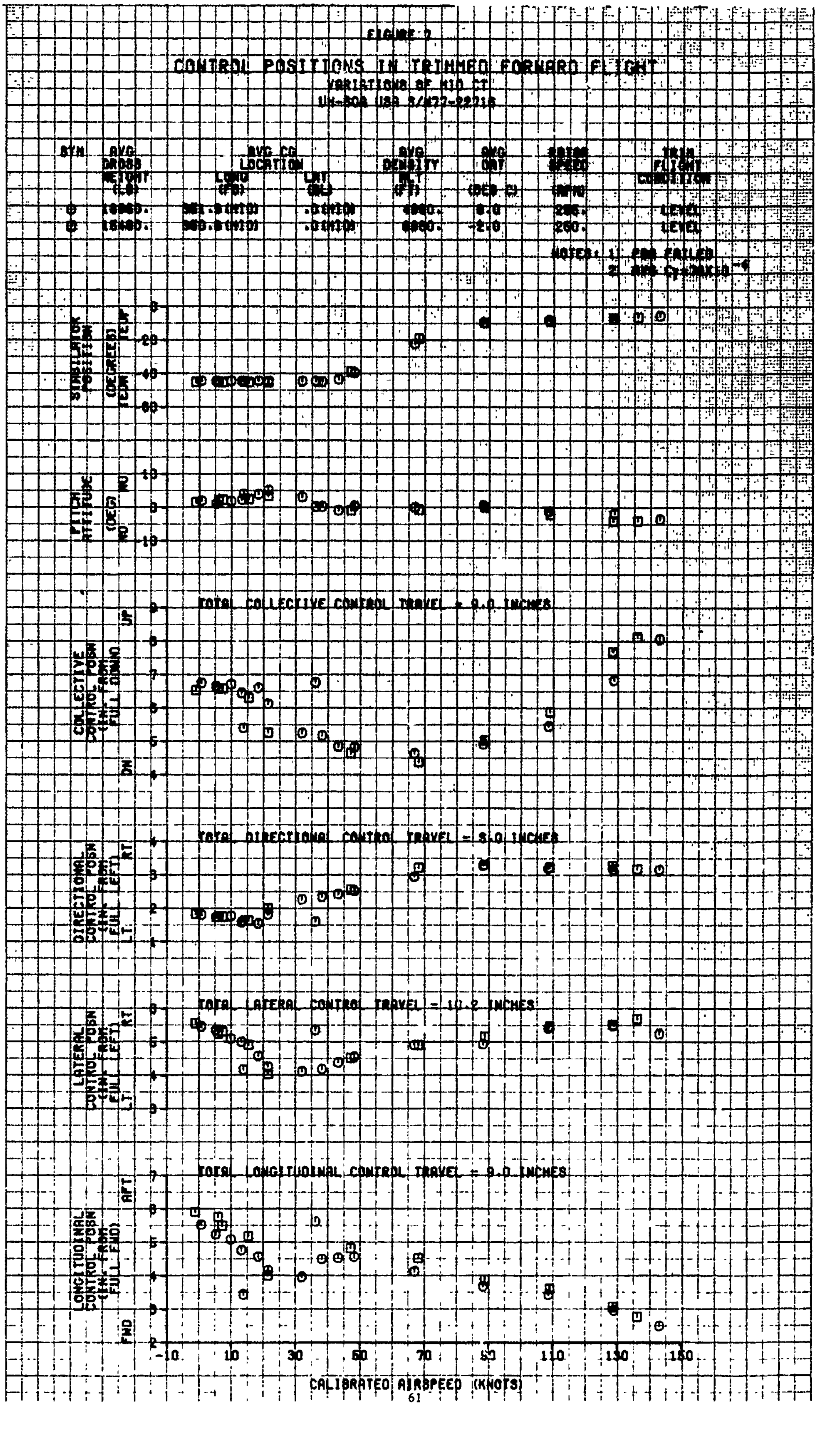


FIGURE 7
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

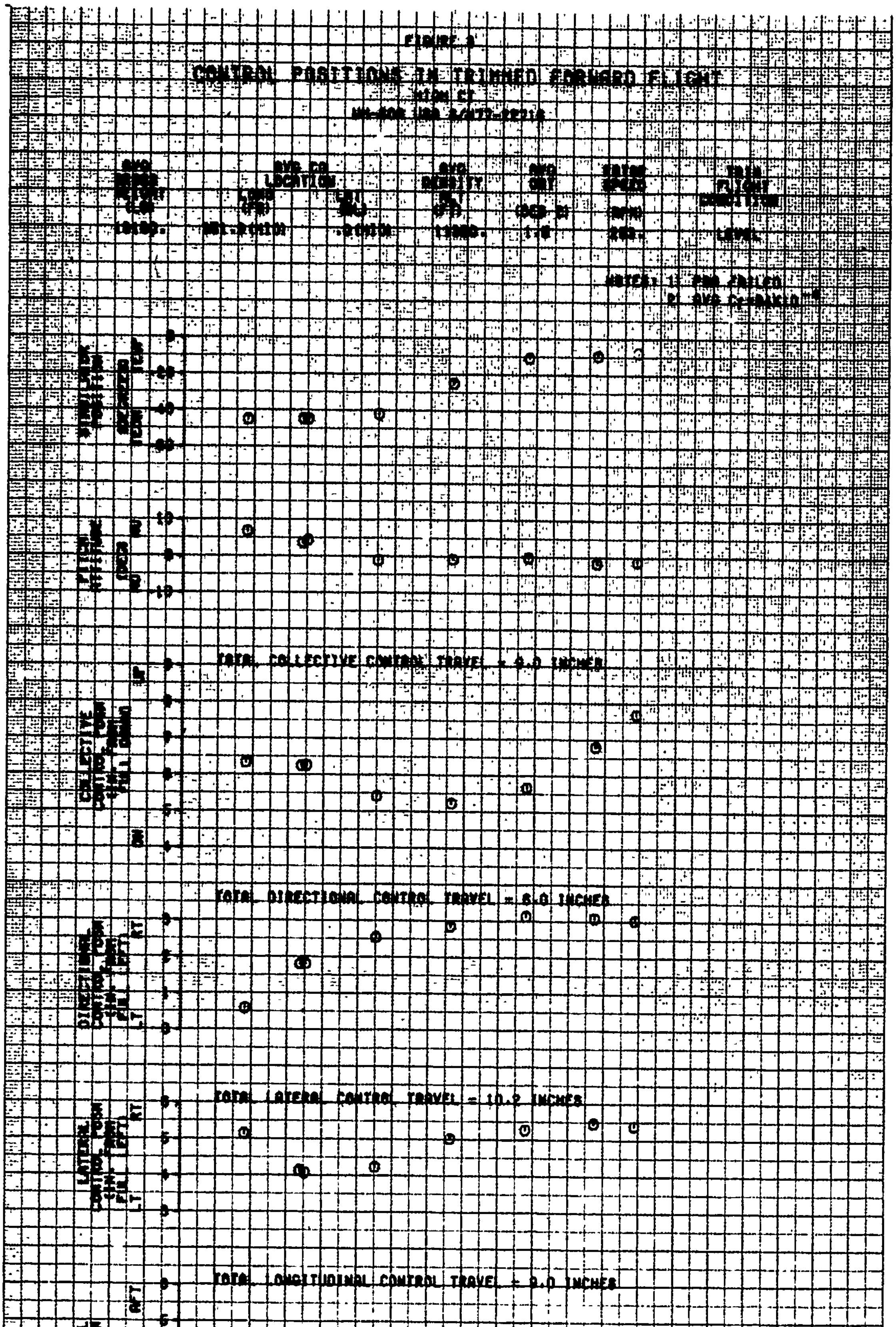
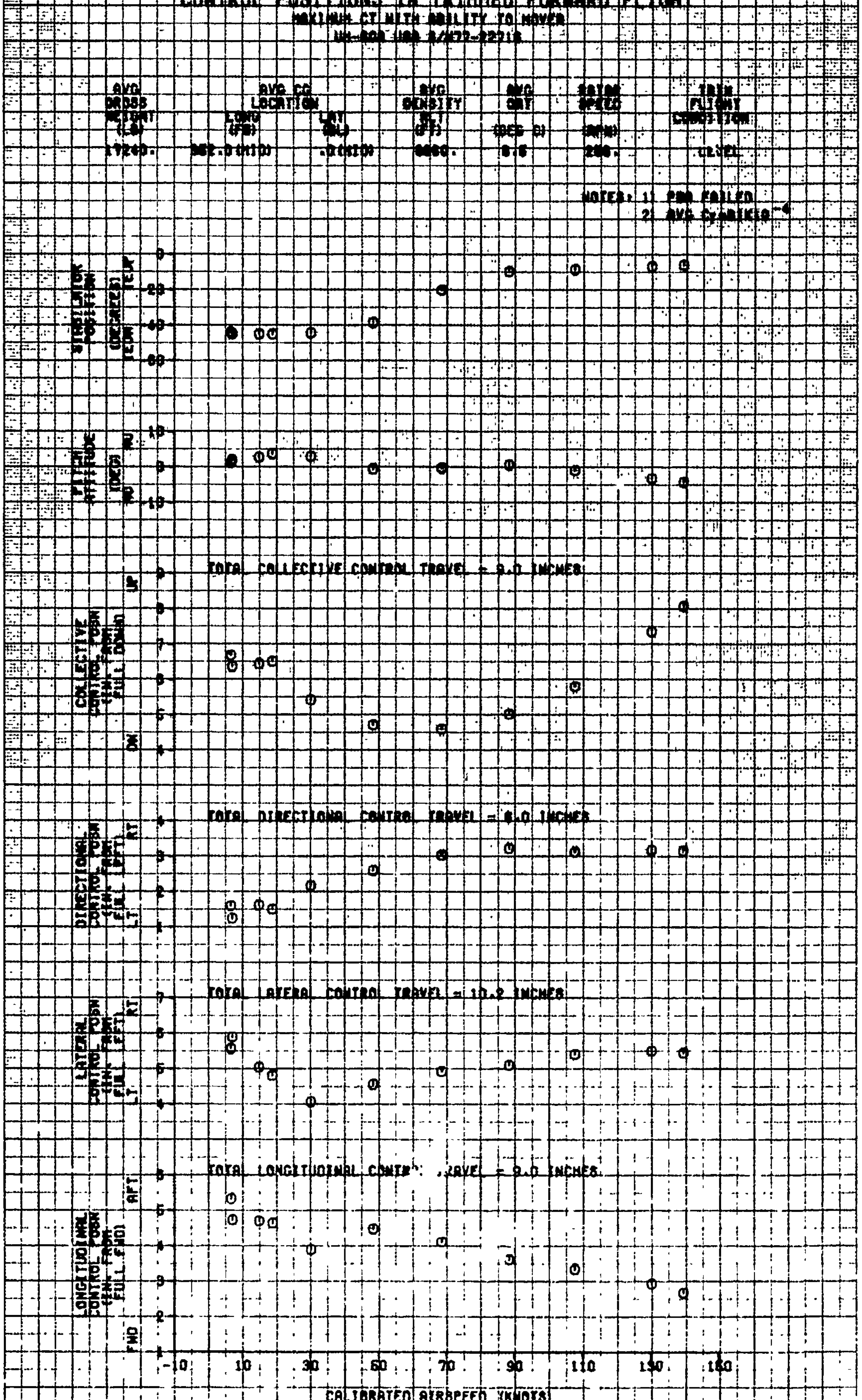
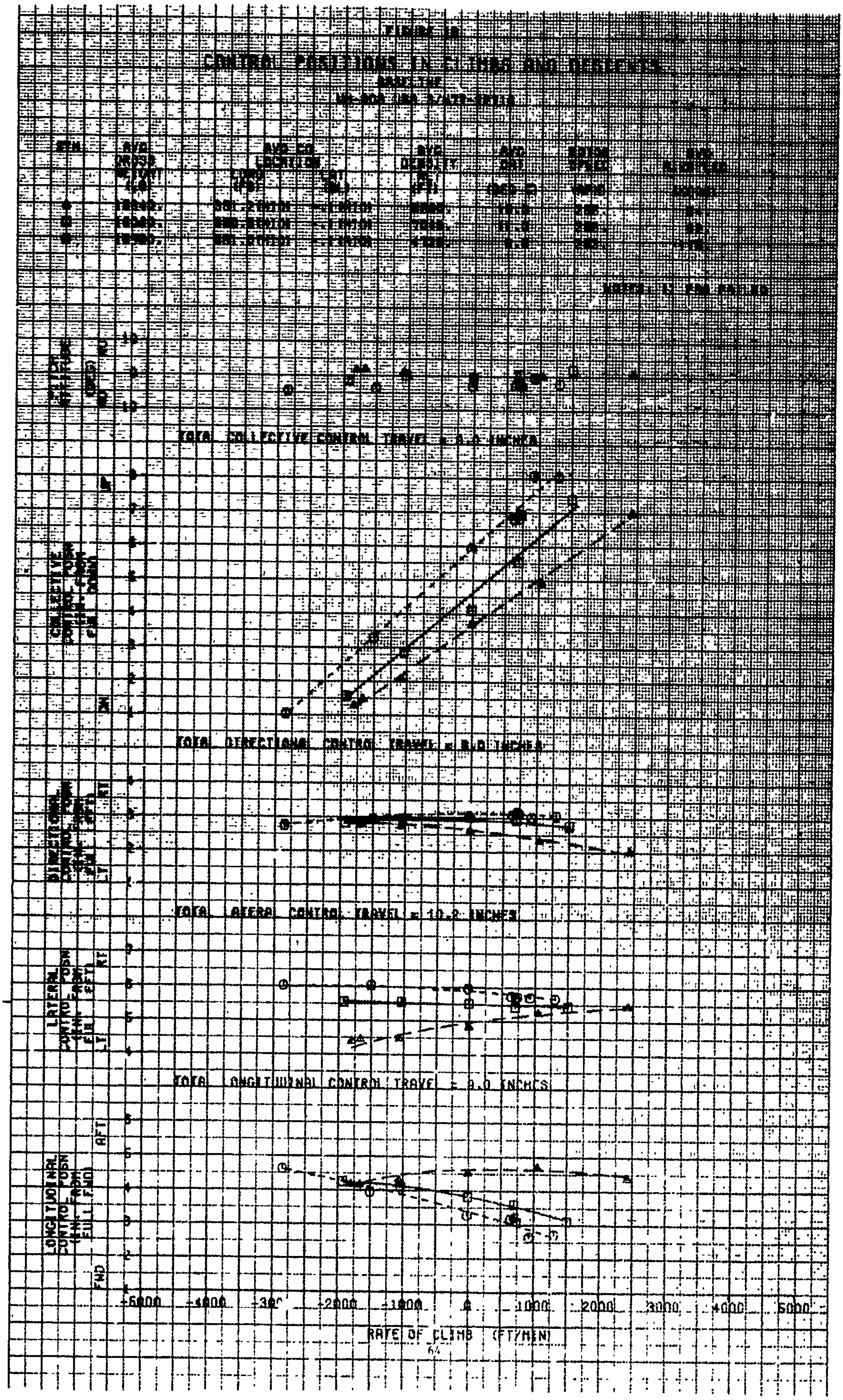

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
LEVEL
TOTAL COLLECTIVE CONTROL TRAVEL = 9.0 INCHES
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 INCHES
TOTAL LATERAL CONTROL TRAVEL = 10.2 INCHES
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.0 INCHES
PITCH ATTITUDE
COLLECTIVE CONTROL POSITION
DIRECTIONAL CONTROL POSITION
LATERAL CONTROL POSITION
LONGITUDINAL CONTROL POSITION
-10
10
30
50
70
90
110
130
150
CALIBRATED AIRSPEED (KNOTS)

FIGURE 9

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT





CONTROL POSITIONS IN CLIMBS AND DESCENTS
TOTAL COLLECTIVE CONTROL TRAVEL = 9.0 INCHES
TOTAL DIRECTIONAL CONTROL TRAVEL = 8.0 INCHES
TOTAL LATERAL CONTROL TRAVEL = 10.2 INCHES
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.0 INCHES
LONGITUDINAL CONTROL POSN (IN. FROM FULL FWD)
AFT
FWD
-5000
-4000
-2000
-1000
0
1000
2000
3000
4000
5000
RATE OF CLIMB (FT/MIN)

FIGURE 11

CONTROL POSITIONS IN CLIMBS AND DESCENTS

AFT CG

UH-60A USA S/N77-22716

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	AVG AIRSPEED (KCAS)
○	16200.	359.8(AFT)	.0(MID)	5240.	[illegible].0	255.	55.
□	16440.	359.5(AFT)	.0(MID)	4880.	[illegible].0	255.	83.

NOTES: 1) FPM FAILED

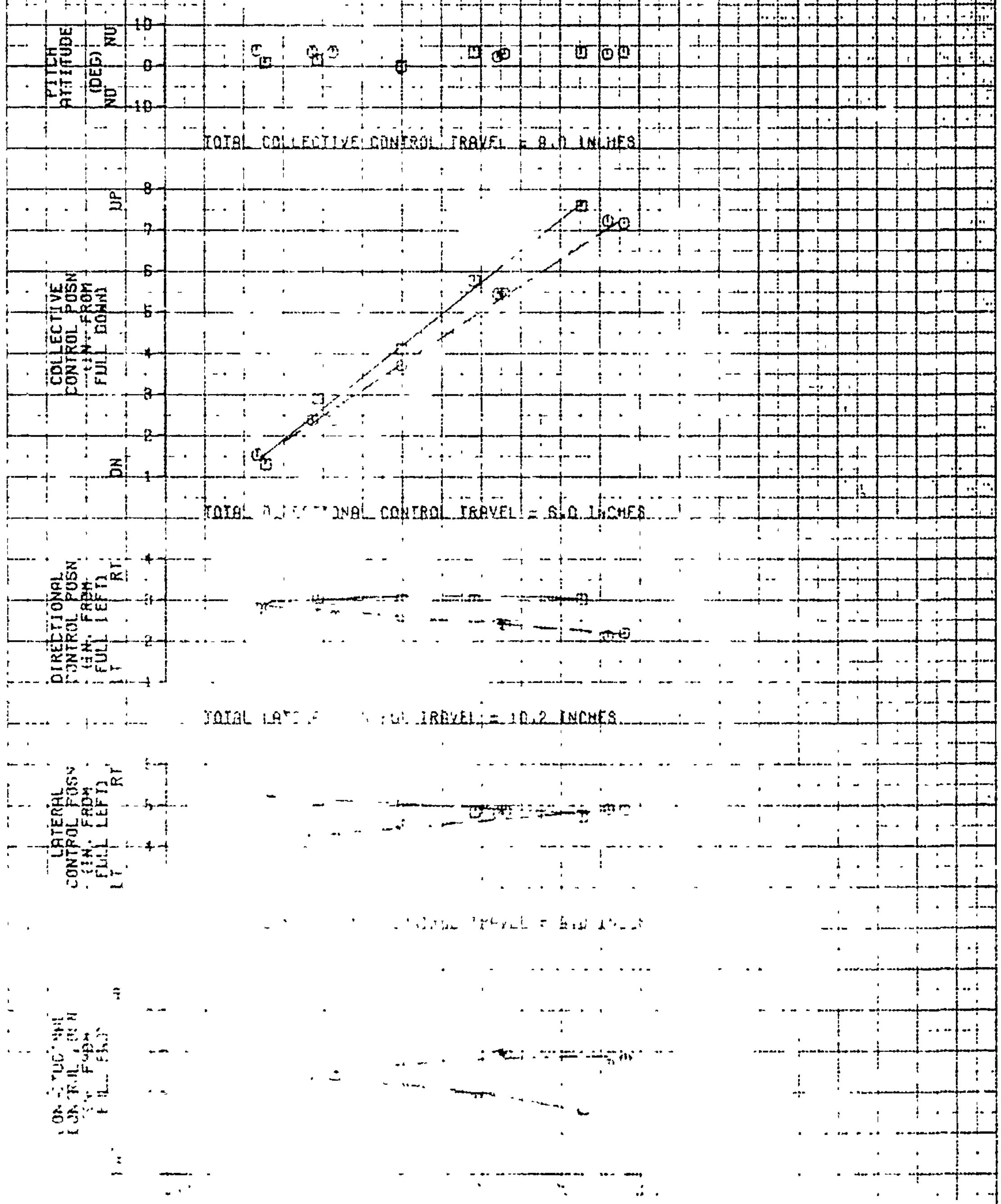


FIGURE 12

COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY

AFT CG- 60 KTS REFERRED TRUE AIRSPEED

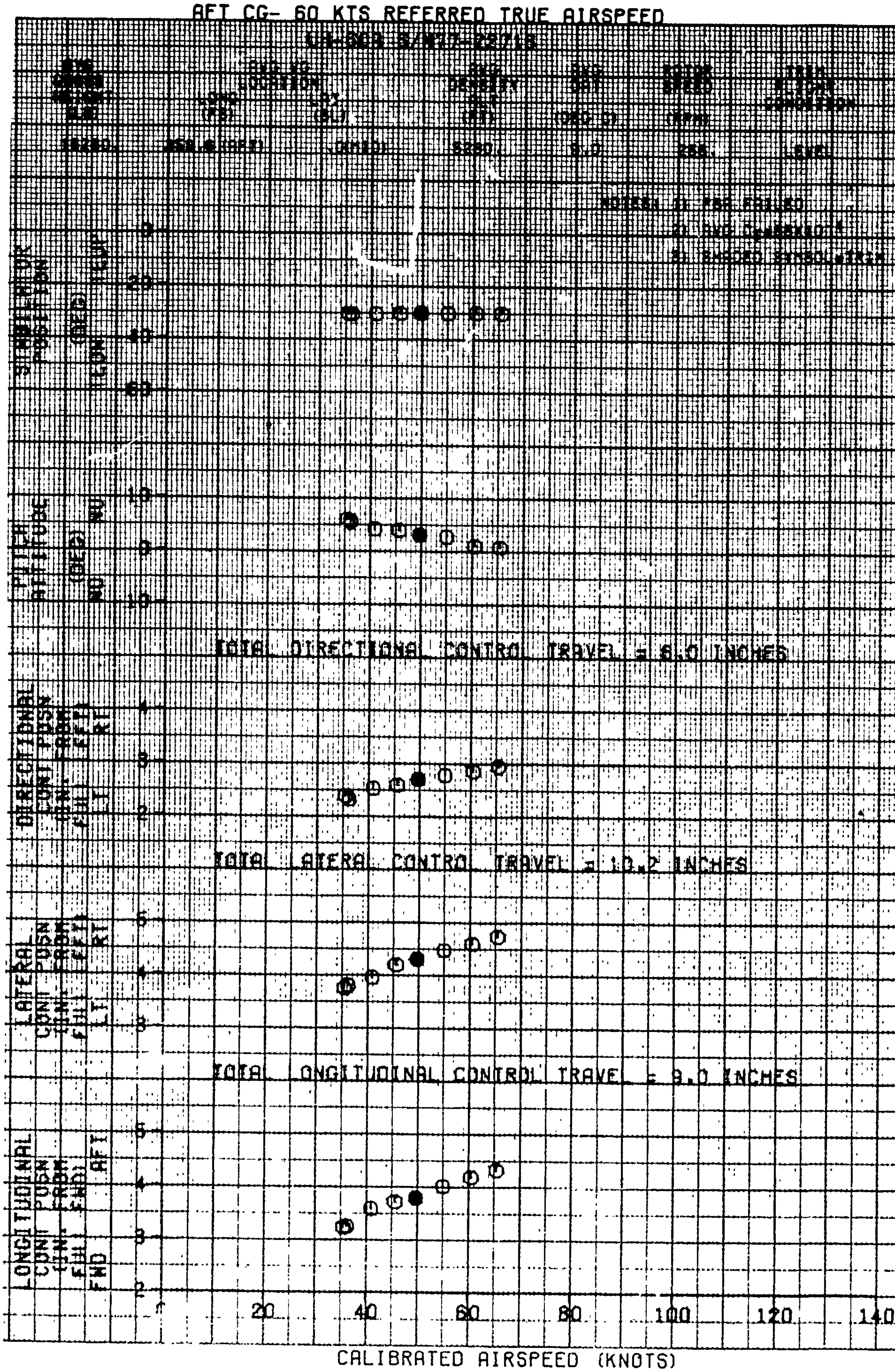


FIGURE 13
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
BASELINE- 60 KTS REFERRED TRUE AIRSPEED
UH-60A S/N77-22716

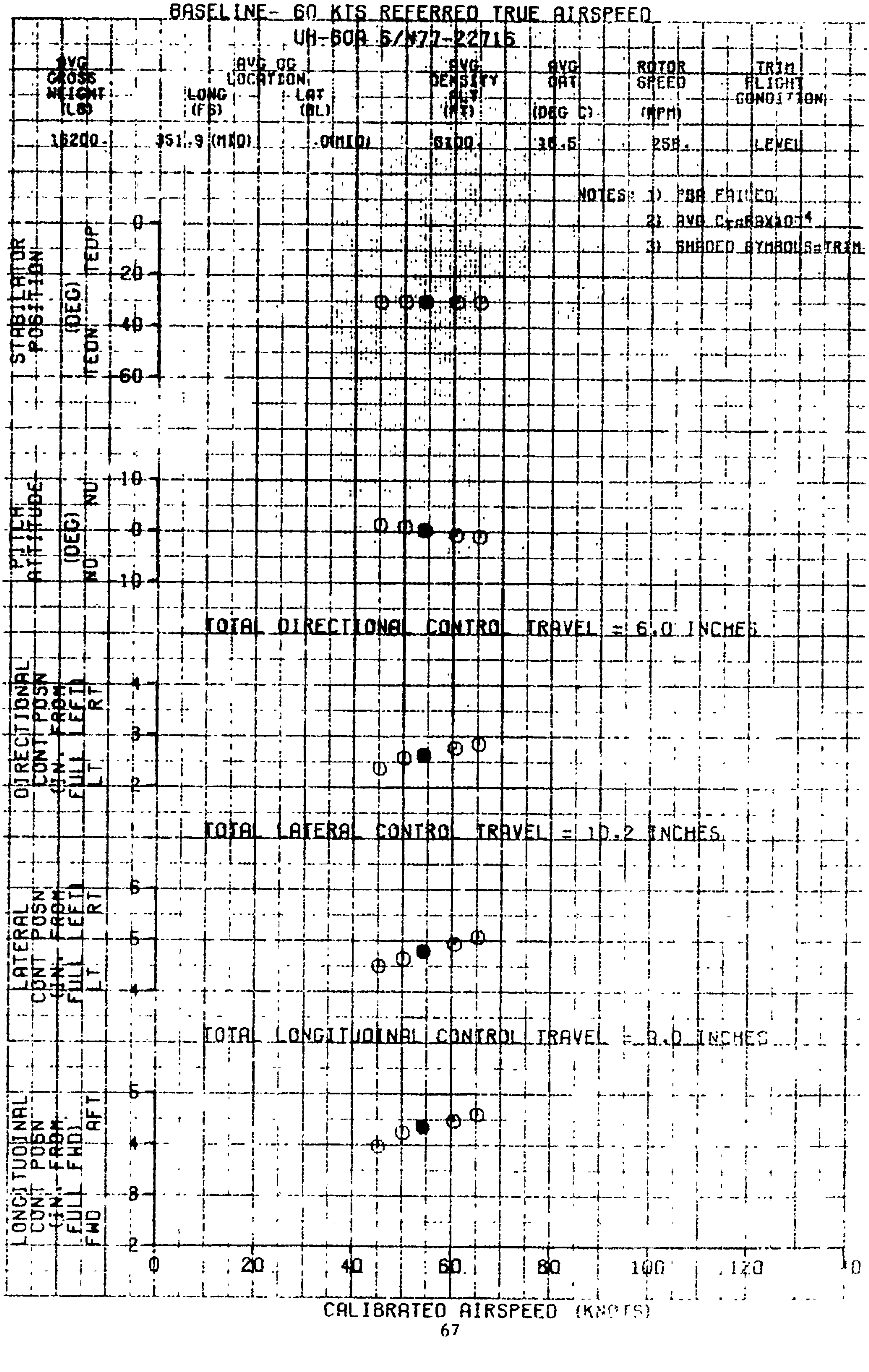


FIGURE 14

COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY

CLIMBS AND DESCENTS

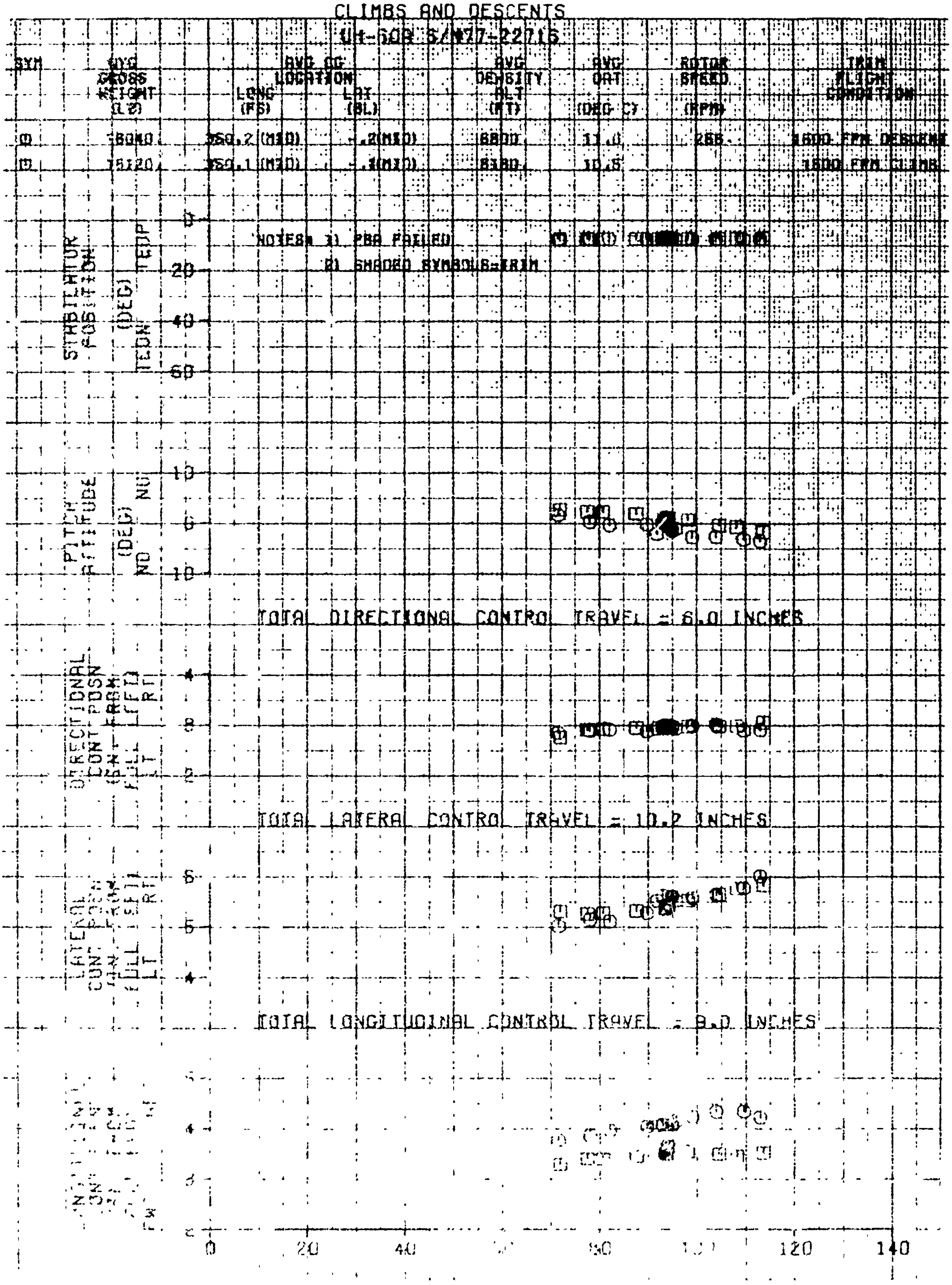


FIGURE 15

COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY

BASELINE-140 KTS REFERRED TRUE AIRSPEED

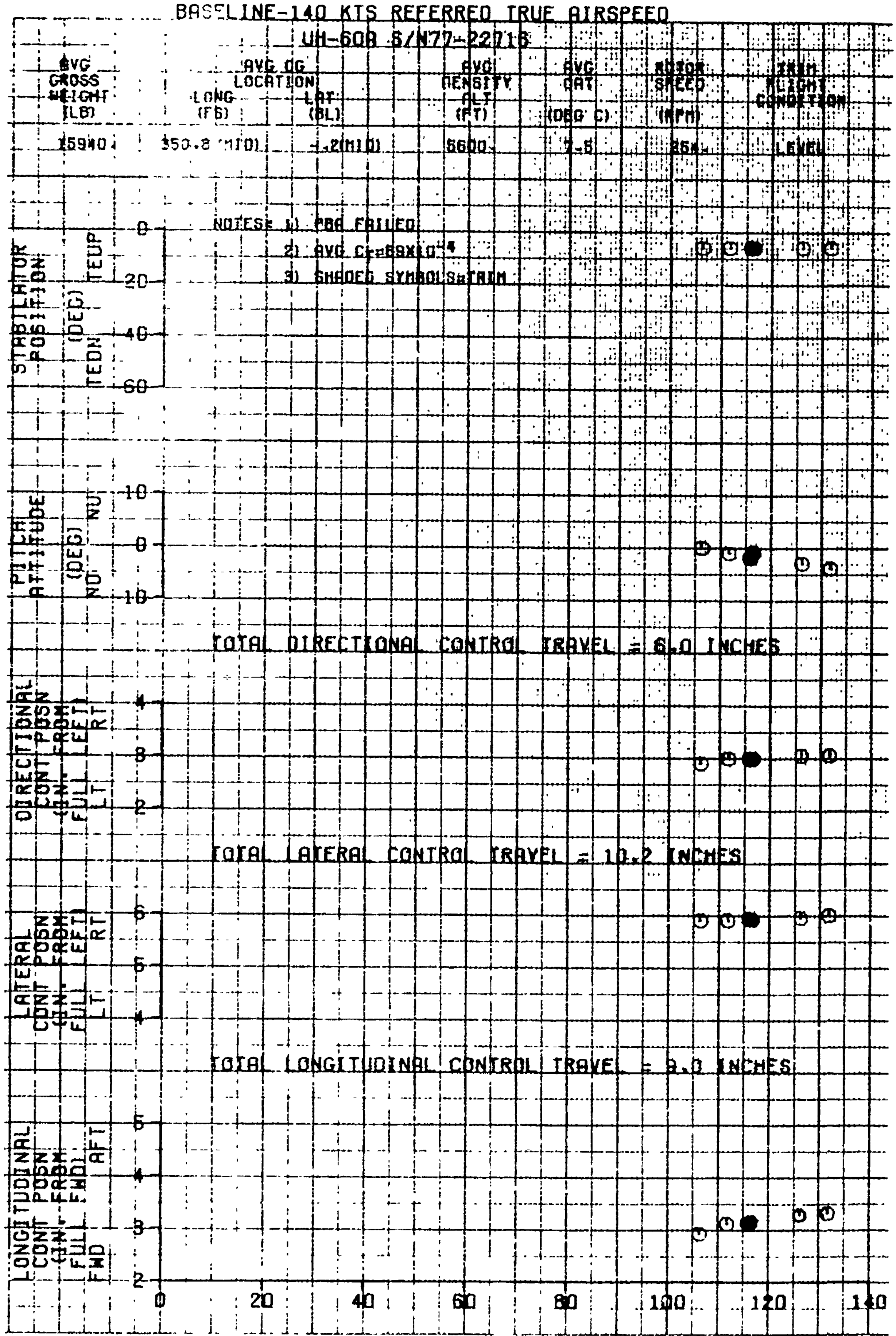


FIGURE 16
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
BASELINE-100 KTS REFERRED TRUE AIRSPEED
UH-60A S/N77-22716

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
16260	351.0 (MID)	.0(MID)	5560.	12.5	257.	LEVEL

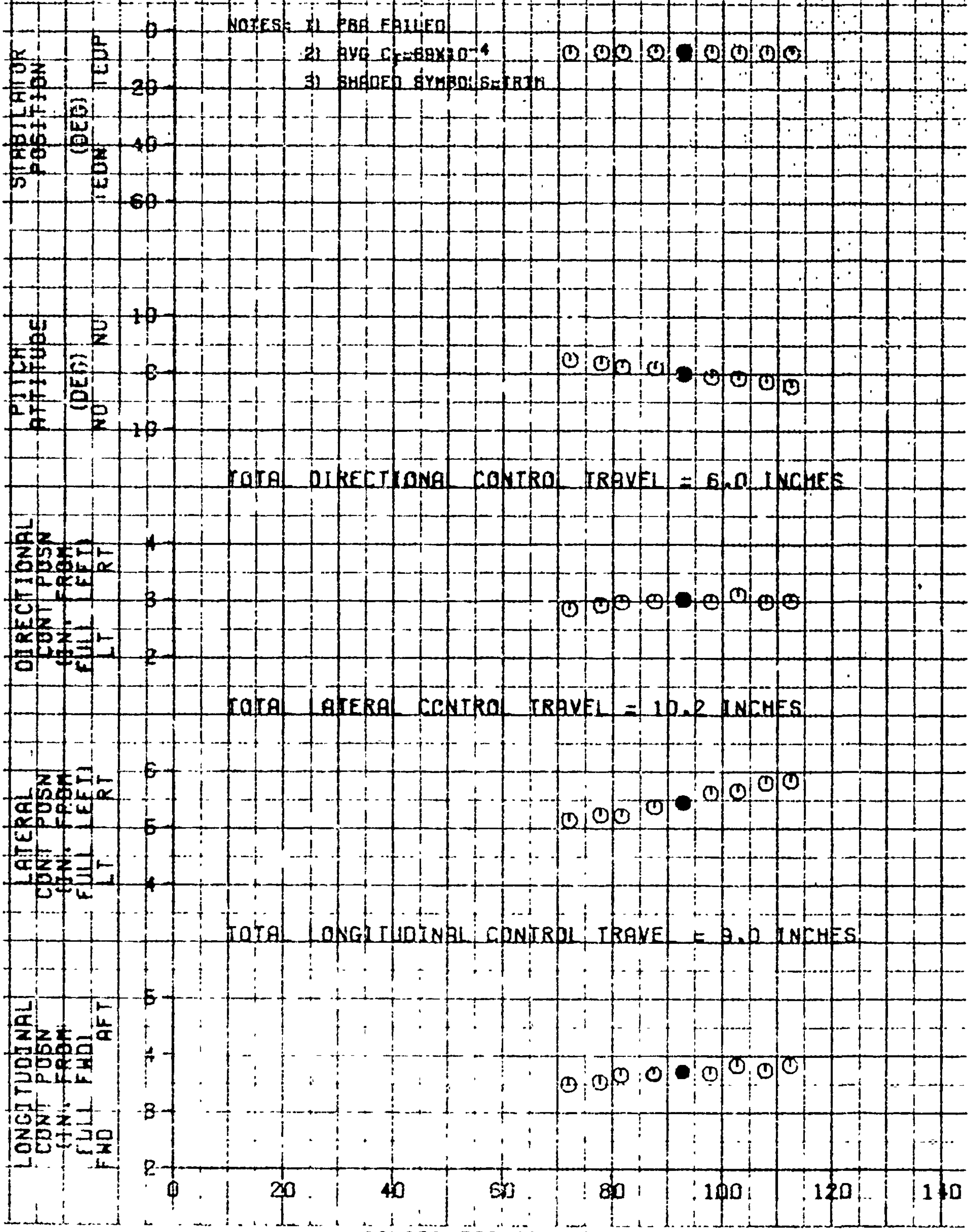


FIGURE 17

COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY

AFT CG-100 KTS REFERRED TRUE AIRSPEED

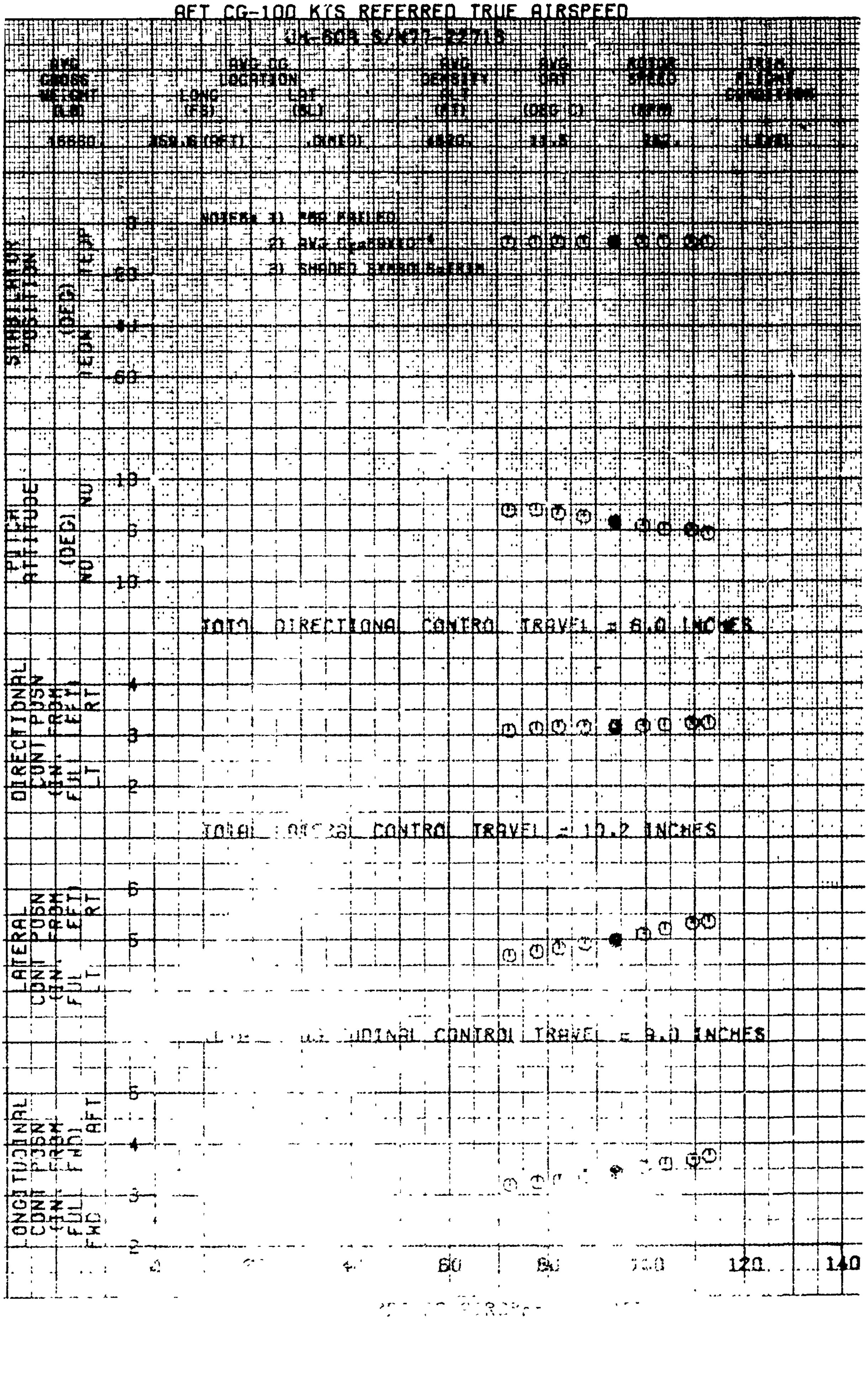


FIGURE 18

COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY

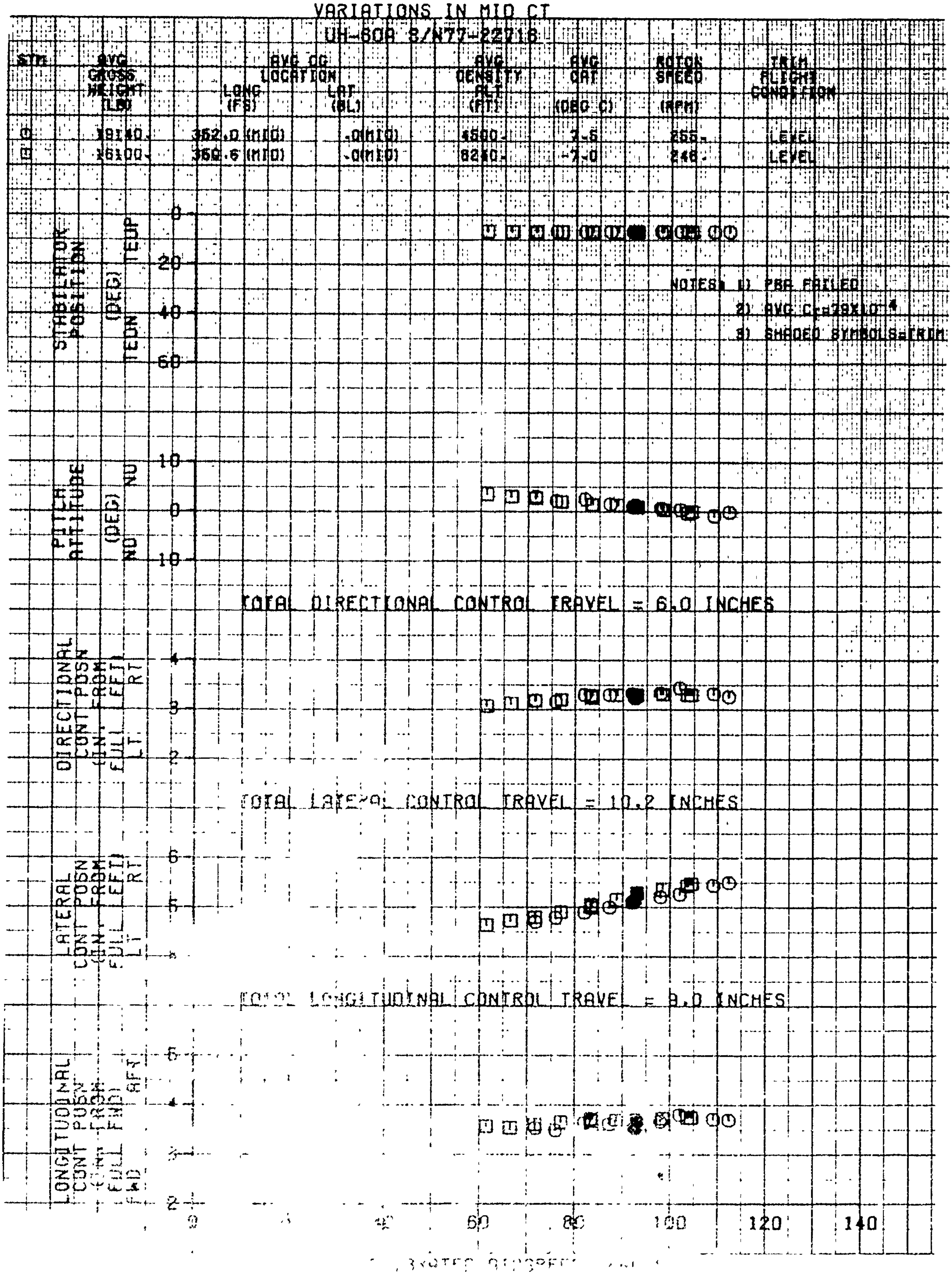
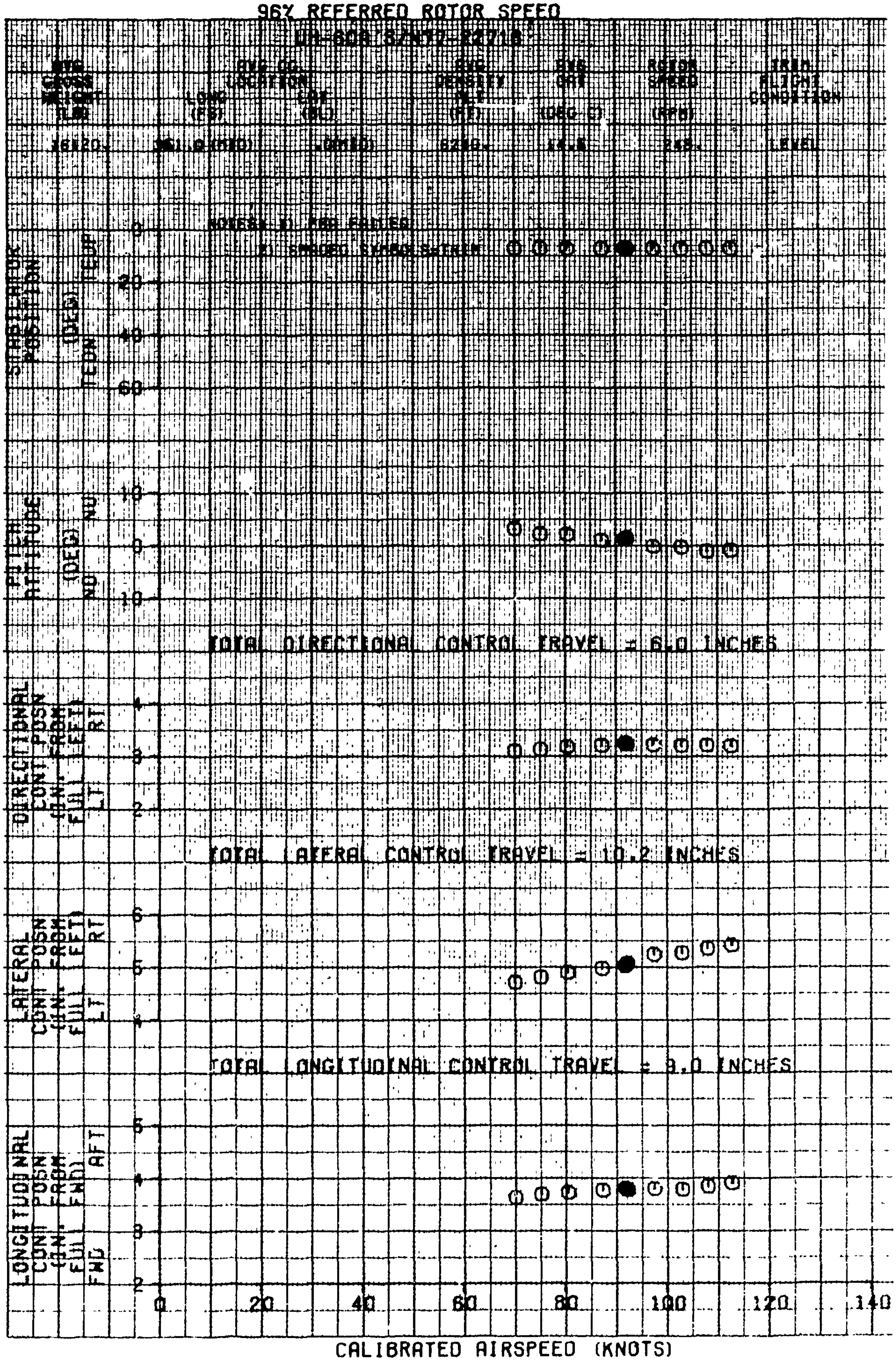
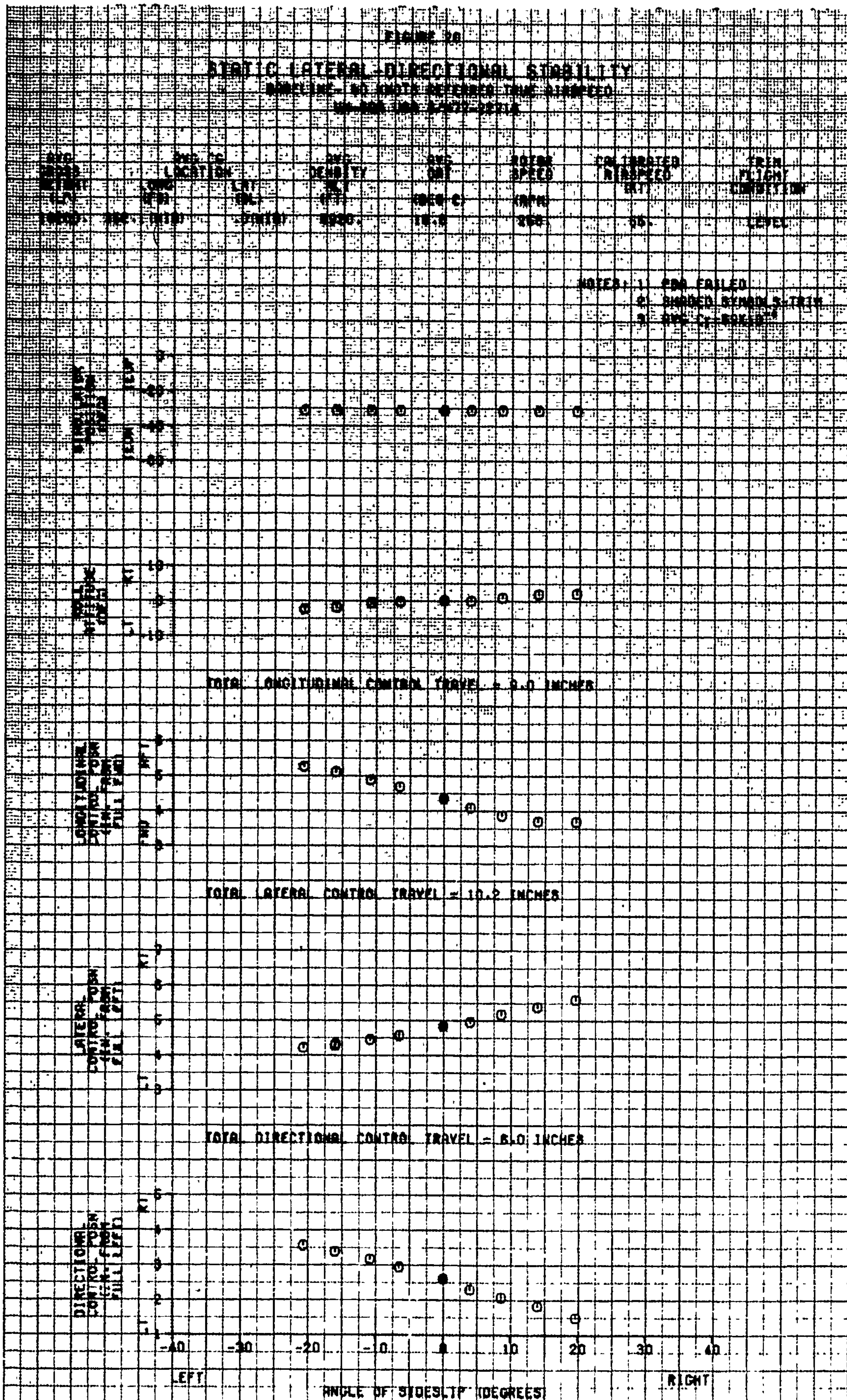


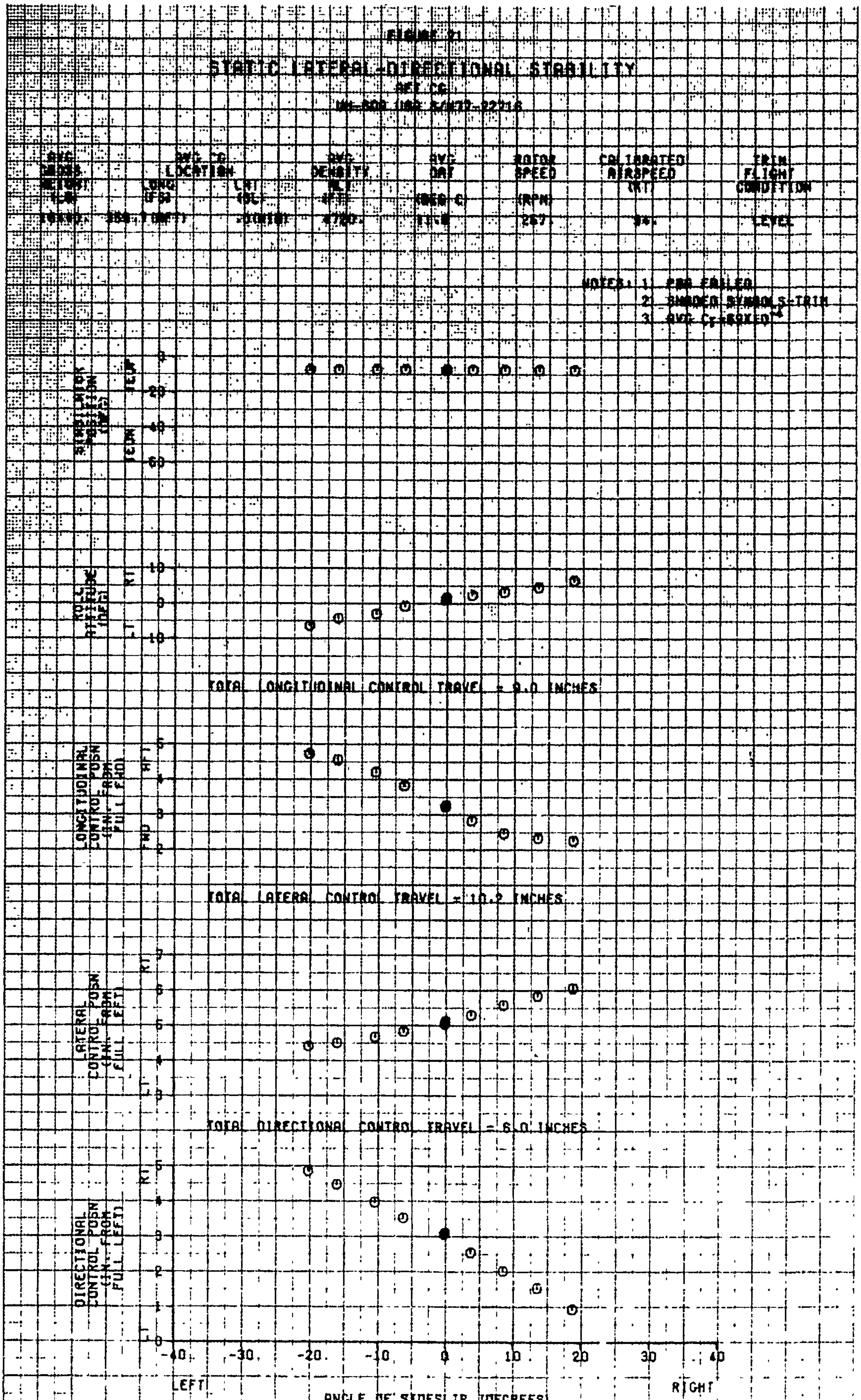
FIGURE 19

COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY

96% REFERRED ROTOR SPEED




STATIC LATERAL-DIRECTIONAL STABILITY
LOCATION
DENSITY
SPEED
AIRSPEED
CONDITION
LEVEL
TOTAL LATERAL CONTROL TRAVEL = 10.2 INCHES
DIRECTIONAL
-40
-30
-20
-10
0
10
20
30
40
LEFT
RIGHT
ANGLE OF SIDESLIP (DEGREES)


STATIC LATERAL-DIRECTIONAL STABILITY
AFT CG
LOCATION
DENSITY
SPEED
CALIBRATED
AIRSPEED
TRIM
FLIGHT
CONDITION
(RPM)
257
94
LEVEL
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.0 INCHES
TOTAL LATERAL CONTROL TRAVEL = 10.2 INCHES
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 INCHES
DIRECTIONAL CONTROL POSN (IN. FROM FULL LEFT)
-40
-30
-20
-10
0
10
20
30
40
LEFT
RIGHT
ANGLE OF SIDESLIP (DEGREES)

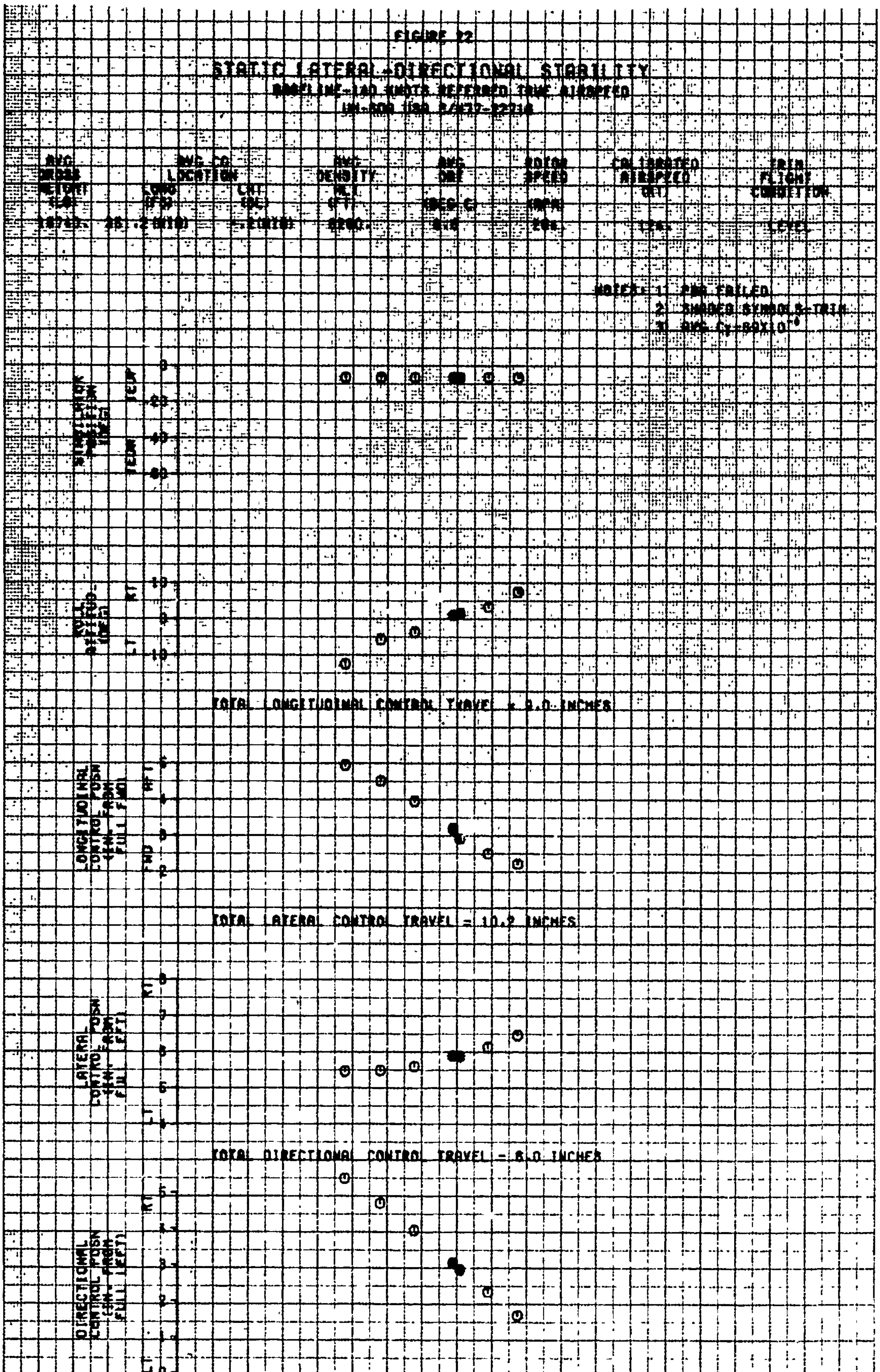


FIGURE 22
STATIC LATERAL-DIRECTIONAL STABILITY
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALT (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
CALIBRATED AIRSPEED (KT)
TRIM FLIGHT CONDITION
LEVEL
NOTES: 1. PMA FAILED
2. SHADED SYMBOLS=TRIM
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.0 INCHES
TOTAL LATERAL CONTROL TRAVEL = 10.2 INCHES
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 INCHES
LATERAL CONTROL POSN (IN. FROM FULL LEFT)
DIRECTIONAL CONTROL POSN (IN. FROM FULL LEFT)
-40
-30
-20
-10
0
10
20
30
40
LEFT
RIGHT
ANGLE OF SIDESLIP (DEGREES)

FIGURE 23

STATIC LATERAL-DIRECTIONAL STABILITY

VARIATIONS OF LATERAL CG

UH-60A USA S/N77-22716

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
○	16460.	361.7(MID)	2.1(RT)	4660.	3.0	262.	92.	LEVEL
□	15760.	361.7(MID)	-4.1(LT)	4600.	12.0	257.	93.	LEVEL

NOTES: 1. PDR FAILED
2. SHADED SYMBOLS=TRIM
3. AVG C_T=88X10^{-4}

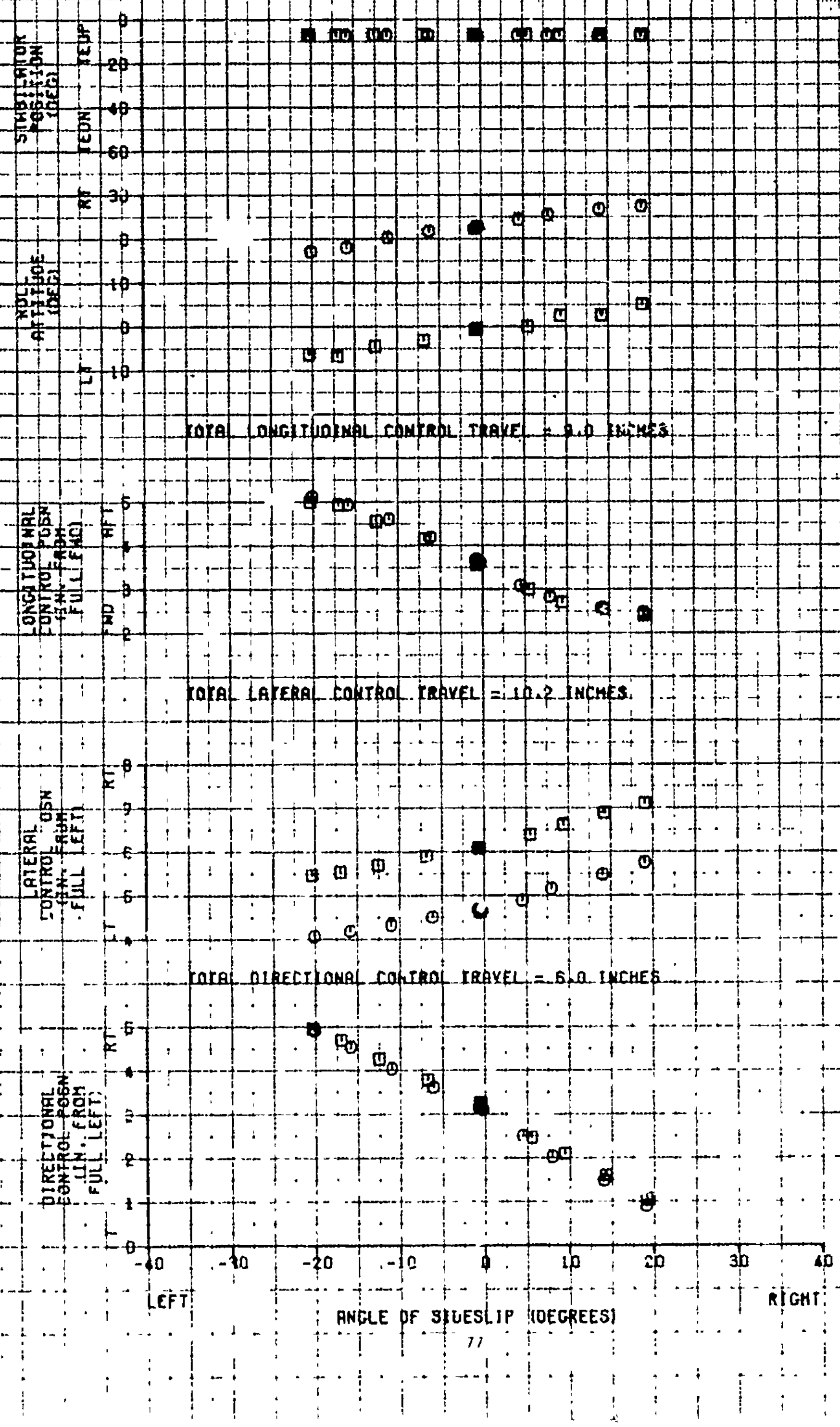


FIGURE 24

STATIC LATERAL-DIRECTIONAL STABILITY

CLIMBS AND DESCENTS

UH-60A USA S/N77-22716

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
●	15720.	358.7(MID)	-.2(MID)	6020.	10.5	258.	93.	1500 FPM DESCENT
⊡	15640.	358.7(MID)	-.2(MID)	6480.	10.0	258.	94.	1500 FPM CLIMB

NOTES: 1. PBA FAILED

2. SHADED SYMBOLS=TRIM

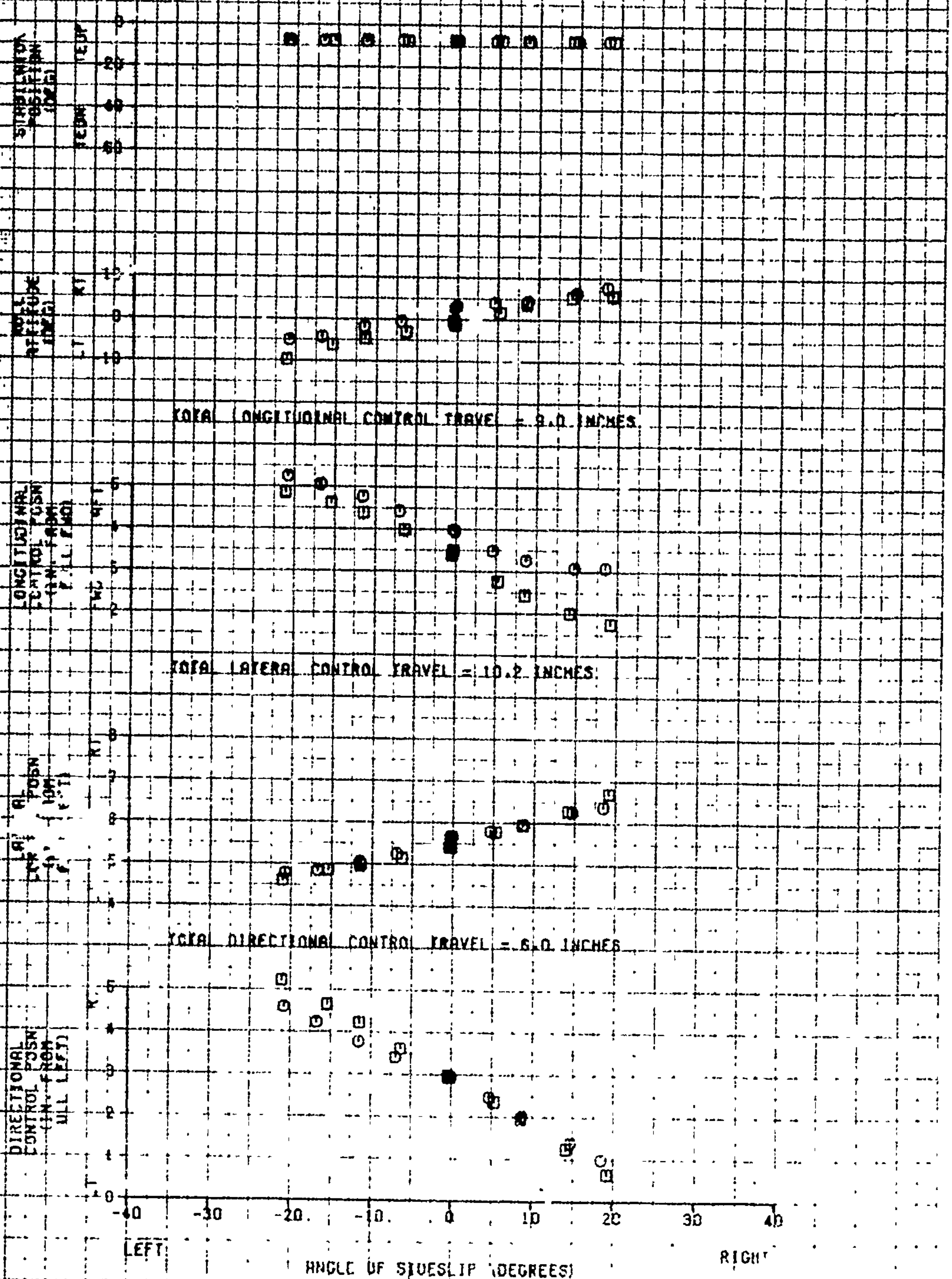
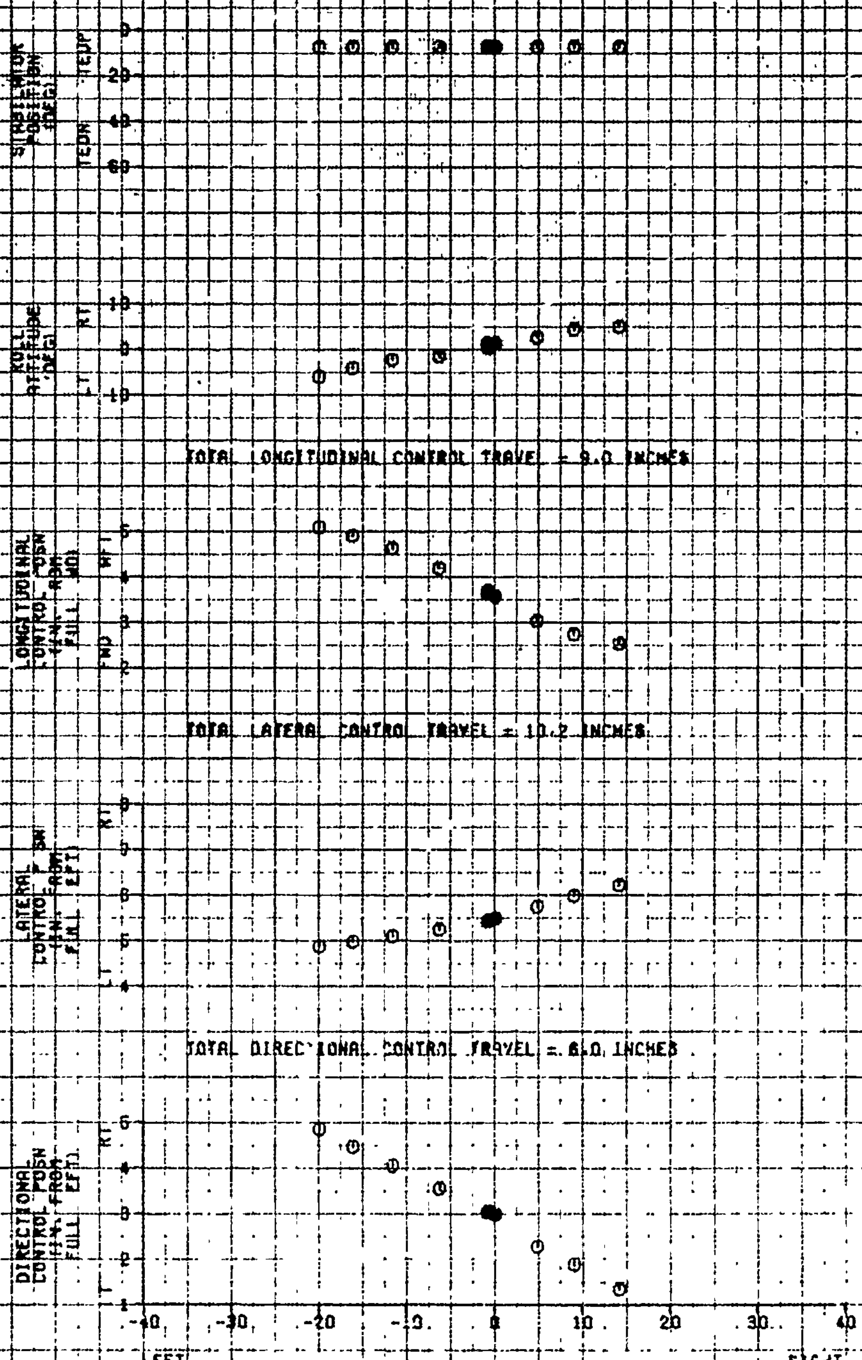


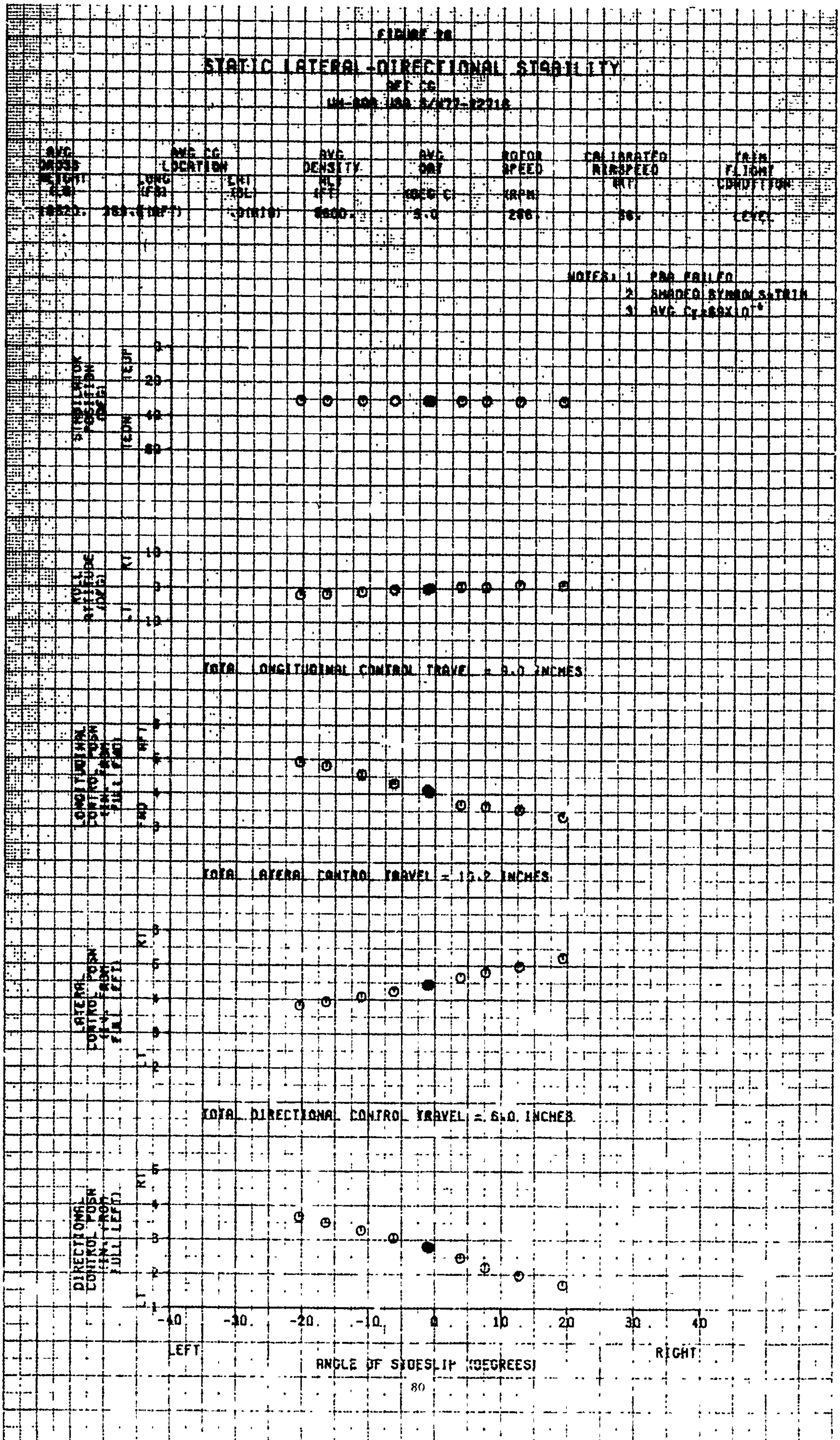
FIGURE 26

STATIC LATERAL-DIRECTIONAL STABILITY
BASELINE-100 KNOTS
UH-60A USA S/N77-22716

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION
16120.	351.2 (MID)	.0(MID)	2020.	12.5	257	92.	LEVEL

NOTES: 1. SAS FAILED
2. SHADED SYMBOL = TRIM
3. AVG C_T = .0089X10^{-4}




STATIC LATERAL-DIRECTIONAL STABILITY
AVG CG LOCATION
AVG DENSITY ALT (FT)
ROTOR SPEED
CALIBRATED AIRSPEED
TRIM FLIGHT CONDITION
NOTES: 1
2
3
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 INCHES
DIRECTIONAL CONTROL POSN (IN. FROM FULL LEFT)
-40
-30
-20
-10
0
10
20
30
40
LEFT
RIGHT
ANGLE OF SIDESLIP (DEGREES)

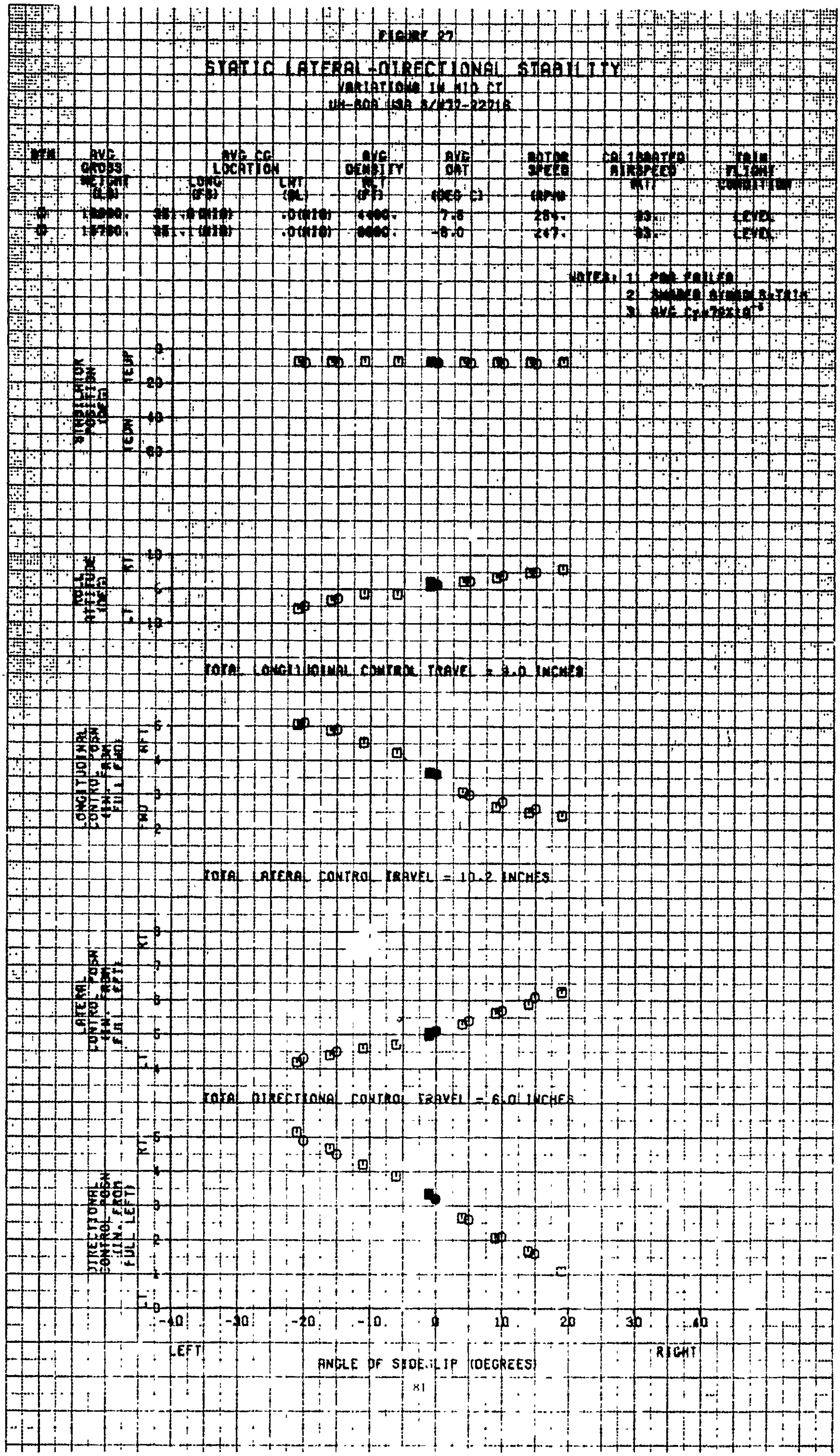
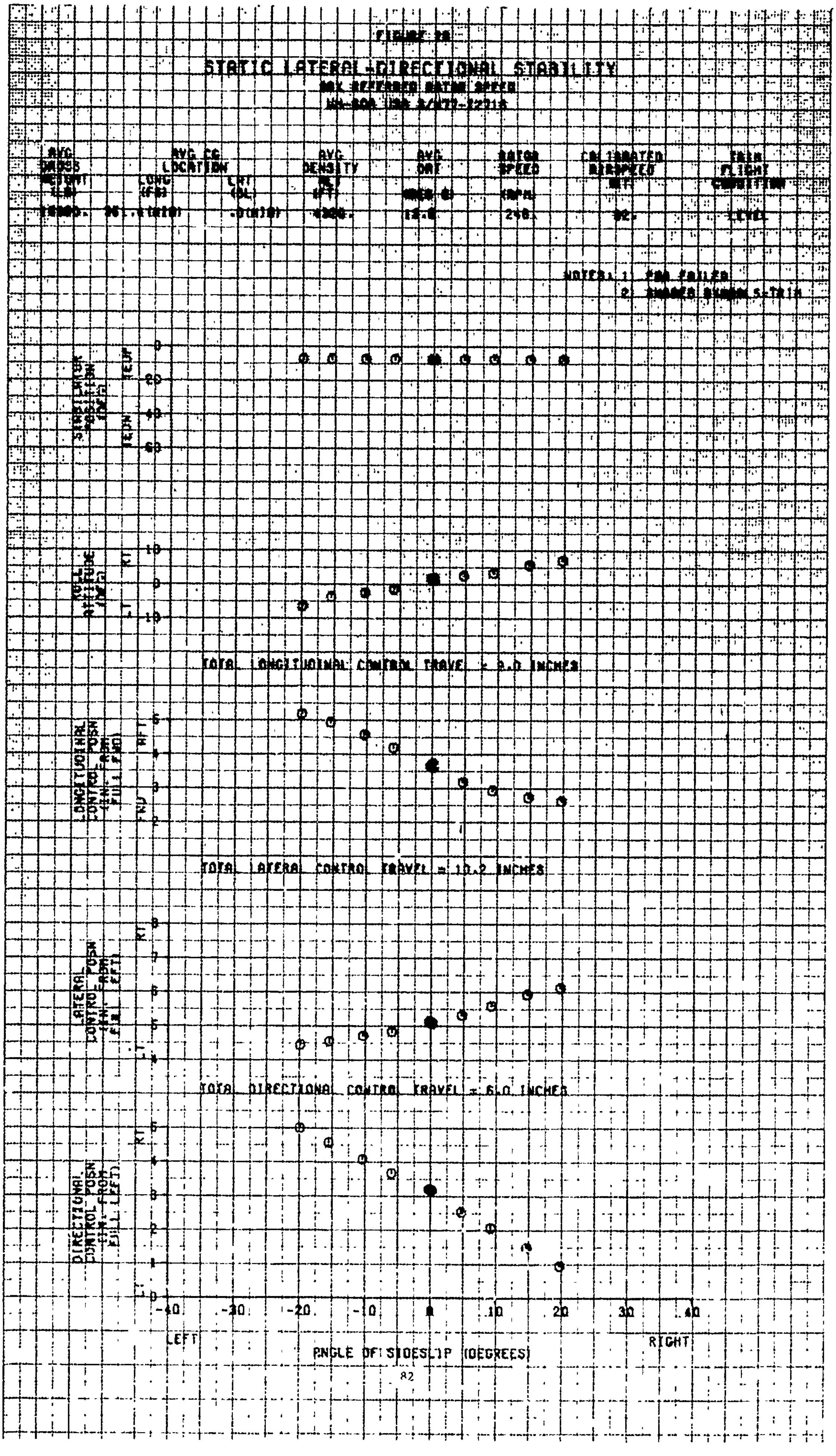
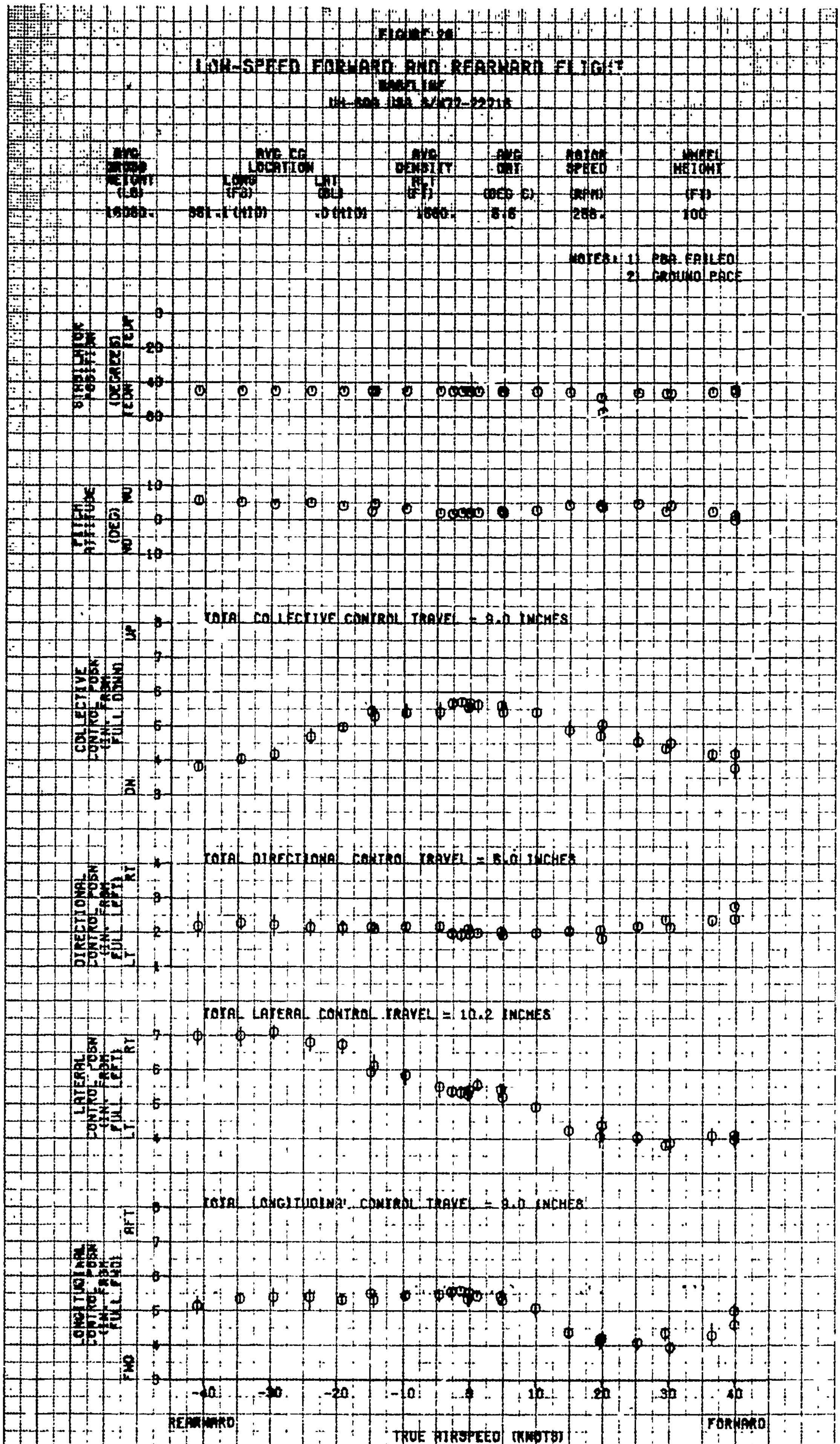


FIGURE 27
STATIC LATERAL-DIRECTIONAL STABILITY
AVG CG
LOCATION
LONG
TOTAL LATERAL CONTROL TRAVEL = 10.2 INCHES
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 INCHES
-40
-30
-20
-10
0
10
20
30
40
LEFT
RIGHT
ANGLE OF SIDESLIP (DEGREES)


STATIC LATERAL-DIRECTIONAL STABILITY
AVG CG LOCATION
LONG
AVG DENSITY
AVG OAT
CALIBRATED AIRSPEED
240.
TOTAL LONGITUDINAL CONTROL TRAVEL
TOTAL LATERAL CONTROL TRAVEL = 10.2 INCHES
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 INCHES
LATERAL CONTROL POSN (IN. FROM FULL LEFT)
DIRECTIONAL CONTROL POSN (IN. FROM FULL LEFT)
-40
-30
-20
-10
0
10
20
30
40
LEFT
RIGHT
ANGLE OF SIDESLIP (DEGREES)


LOW-SPEED FORWARD AND REARWARD FLIGHT
AVG CG LOCATION
DENSITY
ROTOR SPEED
WHEEL HEIGHT
NOTES: 1) PBA FAILED
2) GROUND PACE
PITCH ATTITUDE (DEG)
TOTAL COLLECTIVE CONTROL TRAVEL = 9.0 INCHES
COLLECTIVE CONTROL POSN (IN. FROM FULL DOWN)
TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 INCHES
DIRECTIONAL CONTROL POSN (IN. FROM FULL LEFT)
TOTAL LATERAL CONTROL TRAVEL = 10.2 INCHES
LATERAL CONTROL POSN (IN. FROM FULL LEFT)
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.0 INCHES
LONGITUDINAL CONTROL POSN (IN. FROM FULL FWD)
REARWARD
TRUE AIRSPEED (KNOTS)
FORWARD

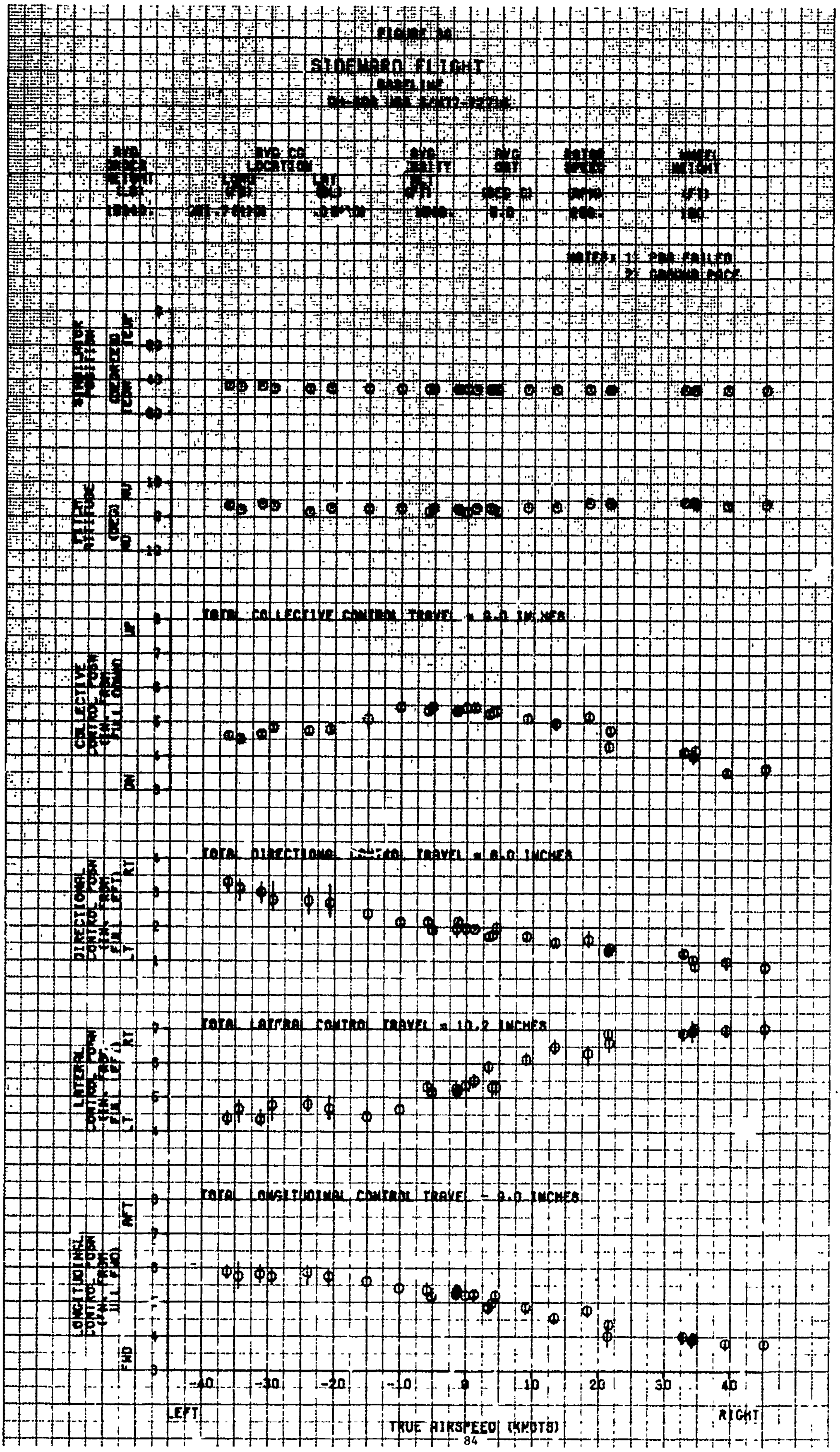

SIDEWARD FLIGHT
TOTAL LATERAL CONTROL TRAVEL = 10.2 INCHES
TOTAL LONGITUDINAL CONTROL TRAVEL = 9.0 INCHES
LEFT
RIGHT
TRUE AIRSPEED (KNOTS)
-40
-30
-20
-10
0
10
20
30
40

FIGURE 81

LOW-SPEED FORWARD AND REARWARD FLIGHT

AFT CG

UH-60A USA S/N77-22716

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	WHEEL HEIGHT (FT)
[illegible]	[illegible] (AFT)	.0 (MID)	[illegible]	[illegible]	[illegible]	100

NOTES: 1) [illegible] FAILED

2) GROUND PACE

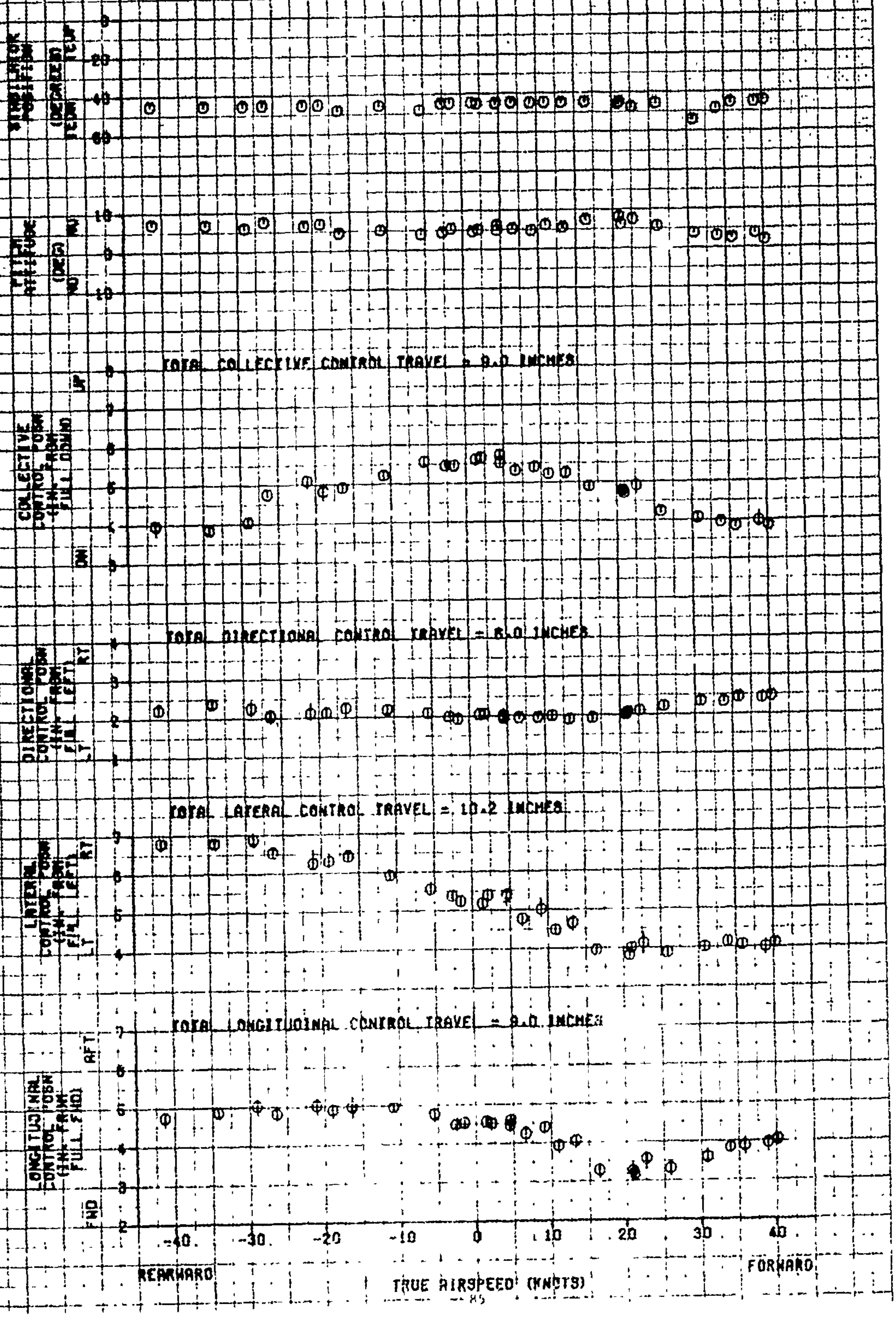


FIGURE 32

SIDEWARD FLIGHT

AFT CG

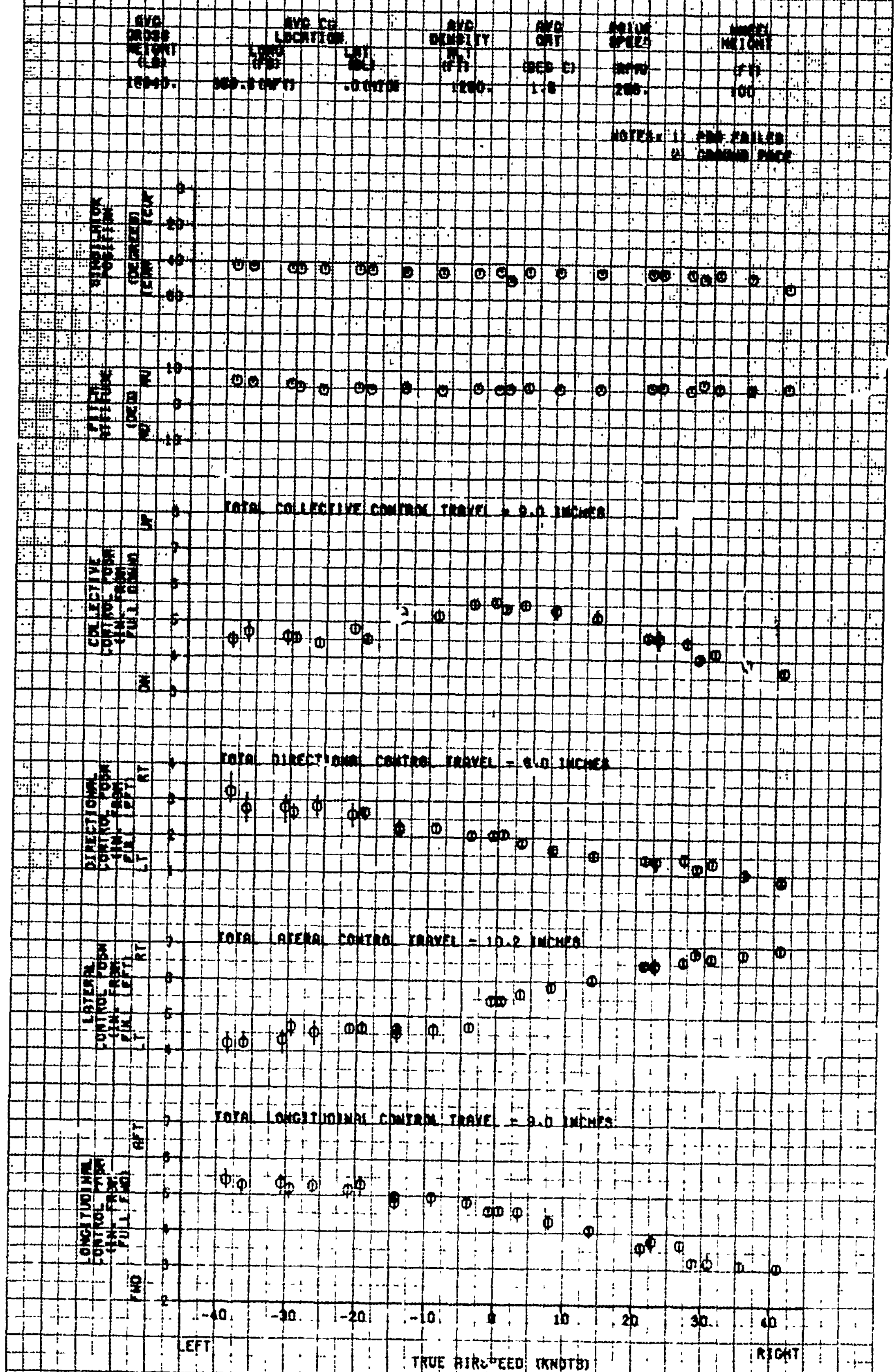


FIGURE 33
STABILATOR SWEEP

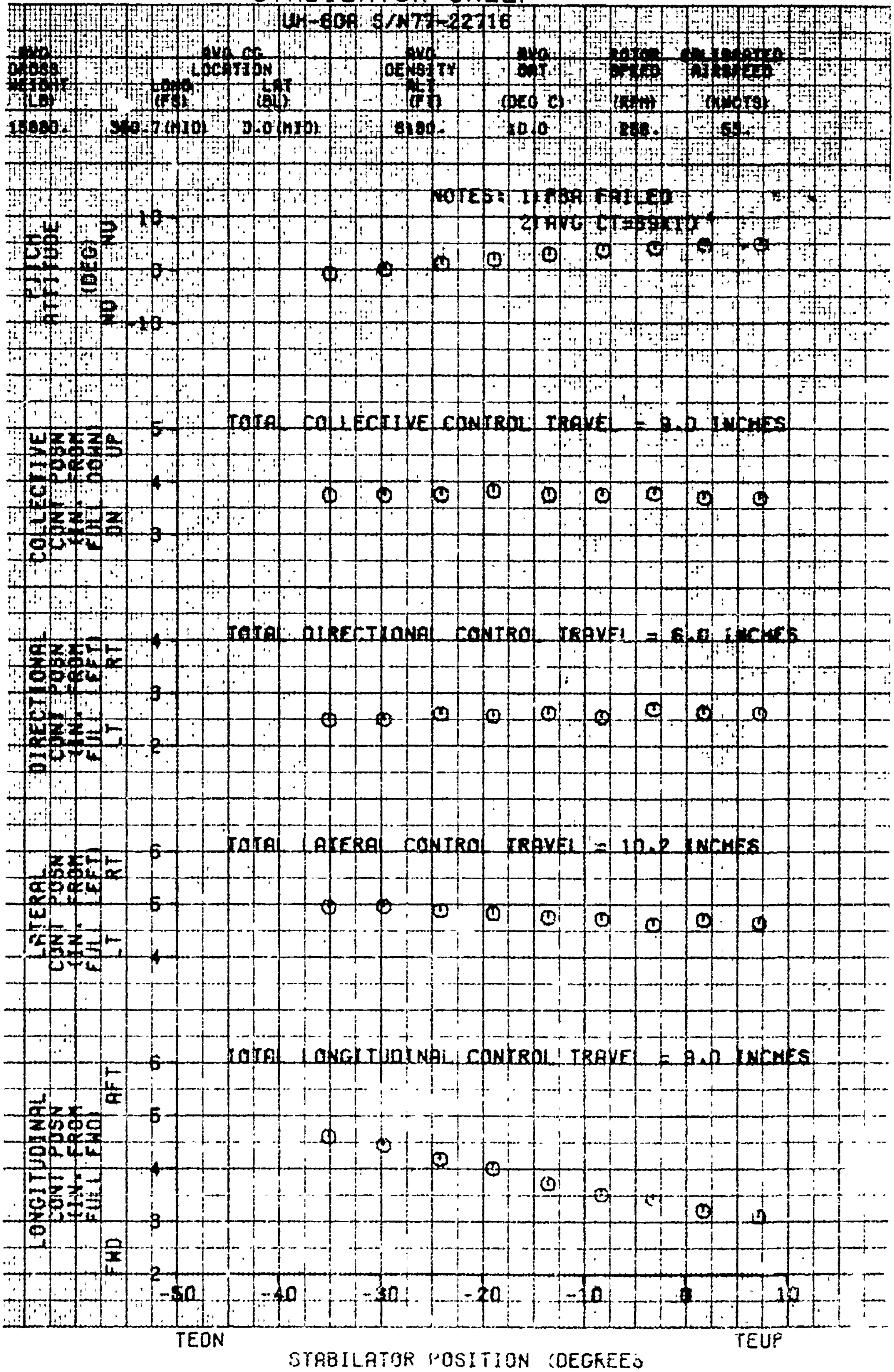


FIGURE 34

STABILATOR SWEEP

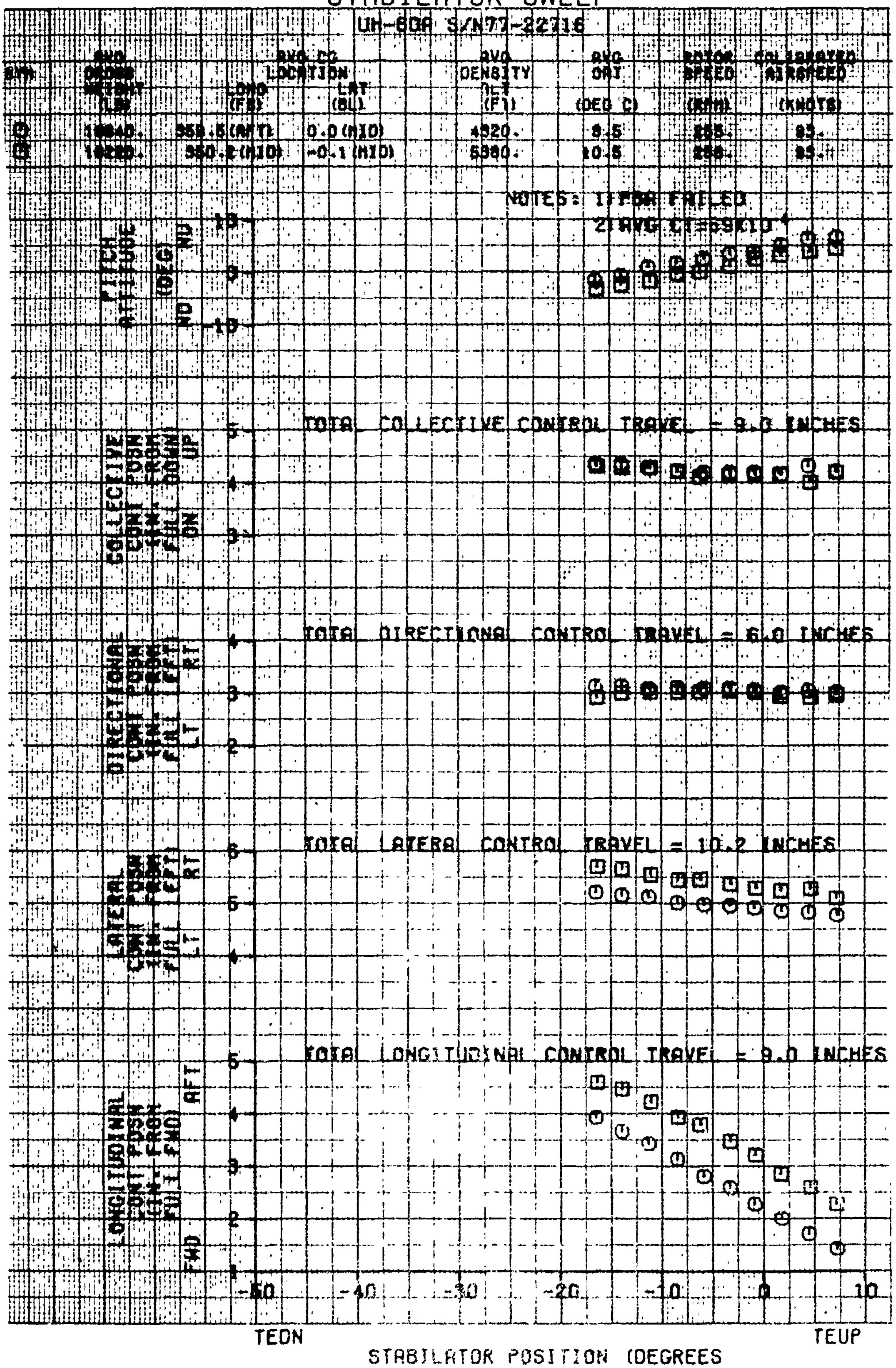
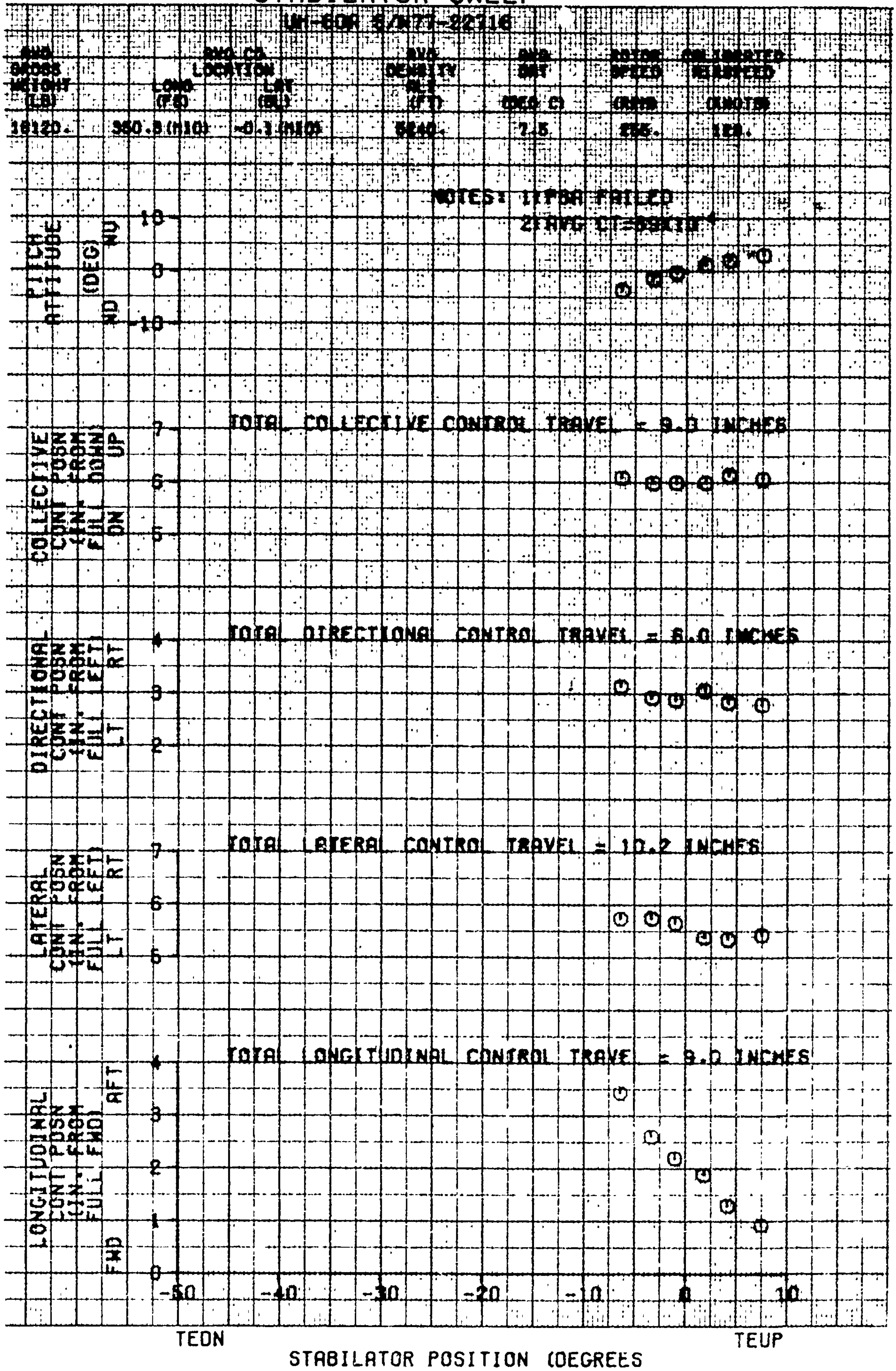


FIGURE 35
STABILATOR SWEEP



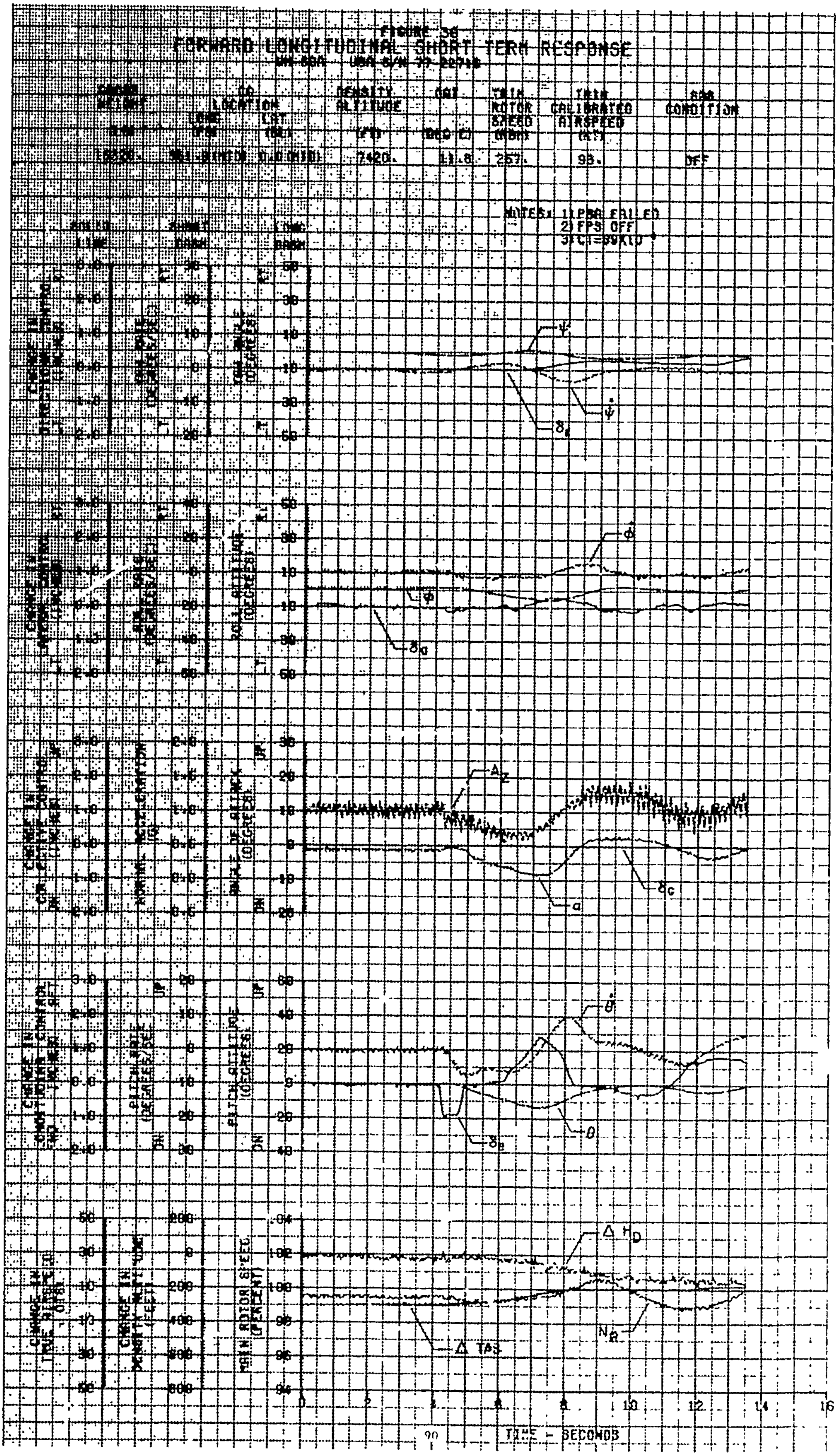
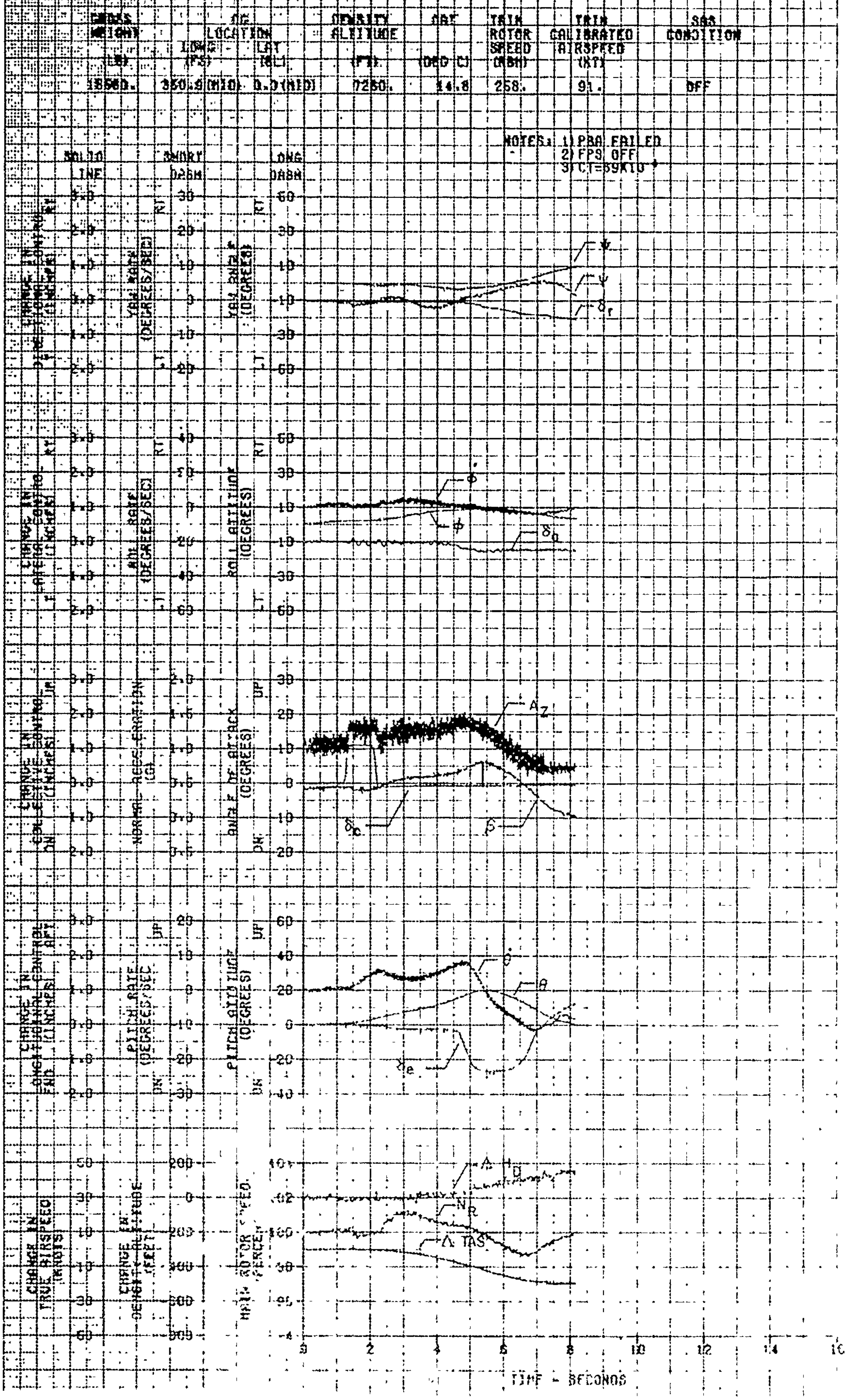


FIGURE 56
FORWARD LONGITUDINAL SHORT TERM RESPONSE
DENSITY ALTITUDE
ROTOR SPEED
CALIBRATED AIRSPEED
CONDITION
7420.
11.8
257.
98.
OFF
NOTES: 1) PBA FAILED
2) FPS OFF
3) CI=S9XID
ROLL ATTITUDE (DEGREES)
ANGLE OF ATTACK (DEGREES)
PITCH ATTITUDE (DEGREES)
PITCH RATE (DEG/SEC)
MAIN ROTOR SPEED (PERCENT)
ψ
$\dot{\psi}$
δ_r
$\dot{\phi}$
ϕ
δ_a
A_Z
α
δ_c
$\dot{\theta}$
θ
δ_e
ΔH_D
Δ TAS
N_R
TIME - SECONDS

FIGURE 37

UP COLLECTIVE SHORT TERM RESPONSE

UH-60A USA S/N 77-22716



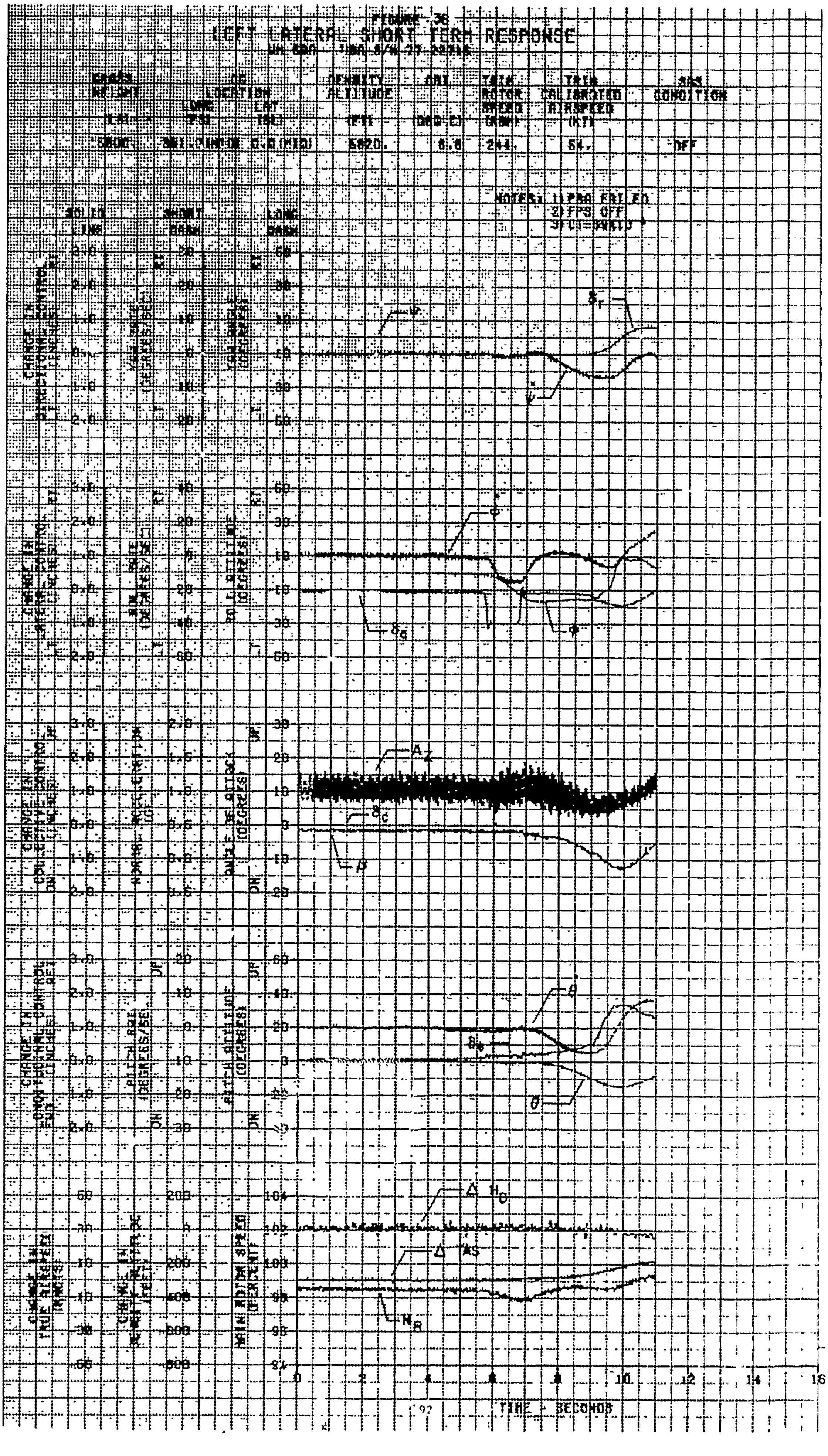


FIGURE 36
LEFT LATERAL SHORT TERM RESPONSE
GROSS WEIGHT
CG LOCATION
LONG
LAT
DENSITY ALTITUDE
(FT)
OAT
TRIM ROTOR SPEED
TRIM CALIBRATED AIRSPEED
(KT)
SCAS CONDITION
5820.
244.
54.
OFF
NOTES: 1) PAS FAILED
2) FPS OFF
SOLID LINE
SHORT DASH
LONG DASH
TIME - SECONDS

FIGURE 39
RIGHT DIRECTIONAL SHORT TERM RESPONSE
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
16000.	351.3(MID)	0.0(MID)	5310.	15.8	258.	55.	OFF

NOTES: 1) PBA FAILED
2) FPS OFF
3) CI=59X10^{-4}

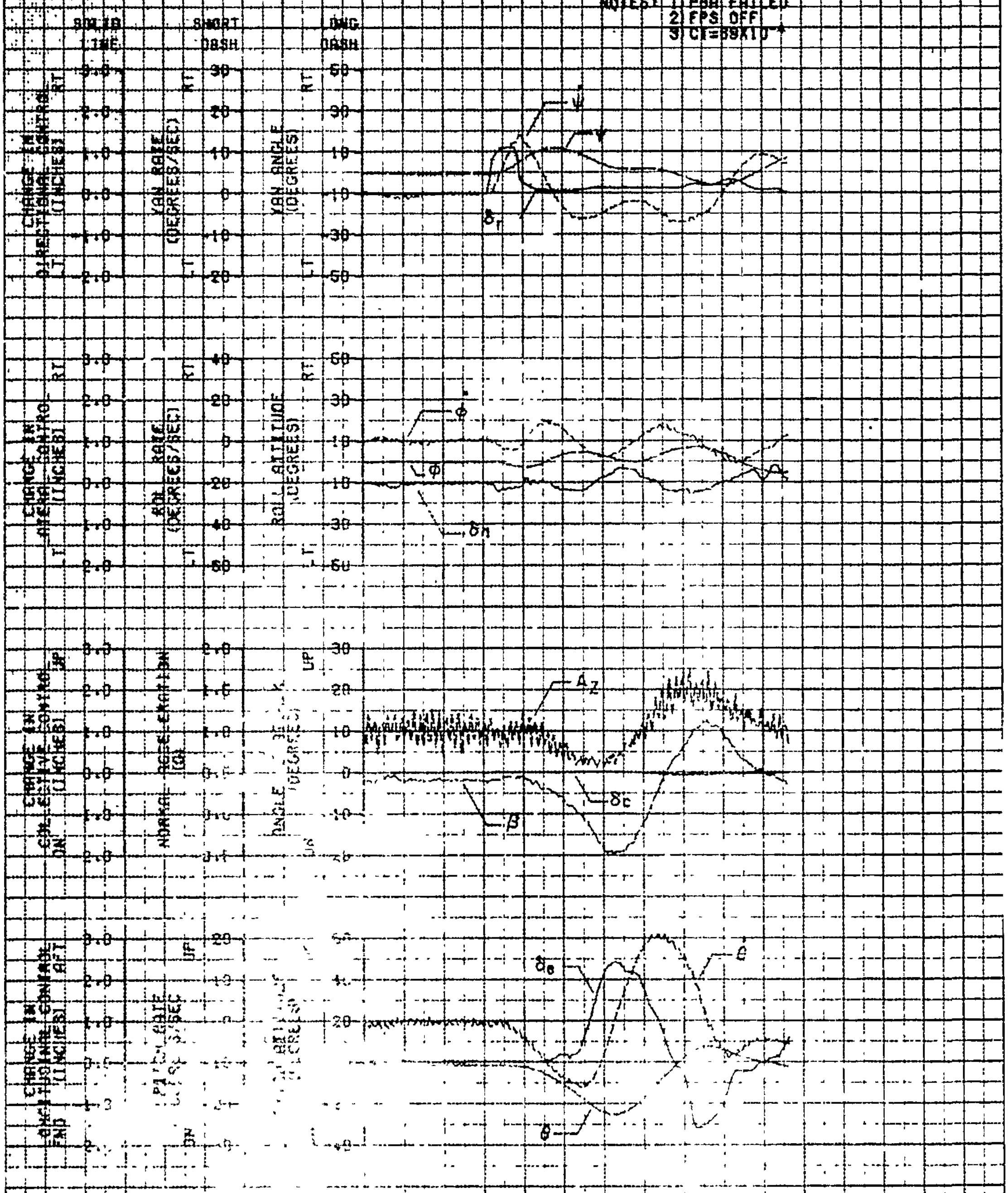


FIGURE 40
LONGITUDINAL LONG TERM RESPONSE
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
15660.	352.0(MID)	0.0(MID)	7320.	16.4	258.	124.	ON

NOTES: 1) PBA FAILED
2) FPS OFF
3) CT=69X10

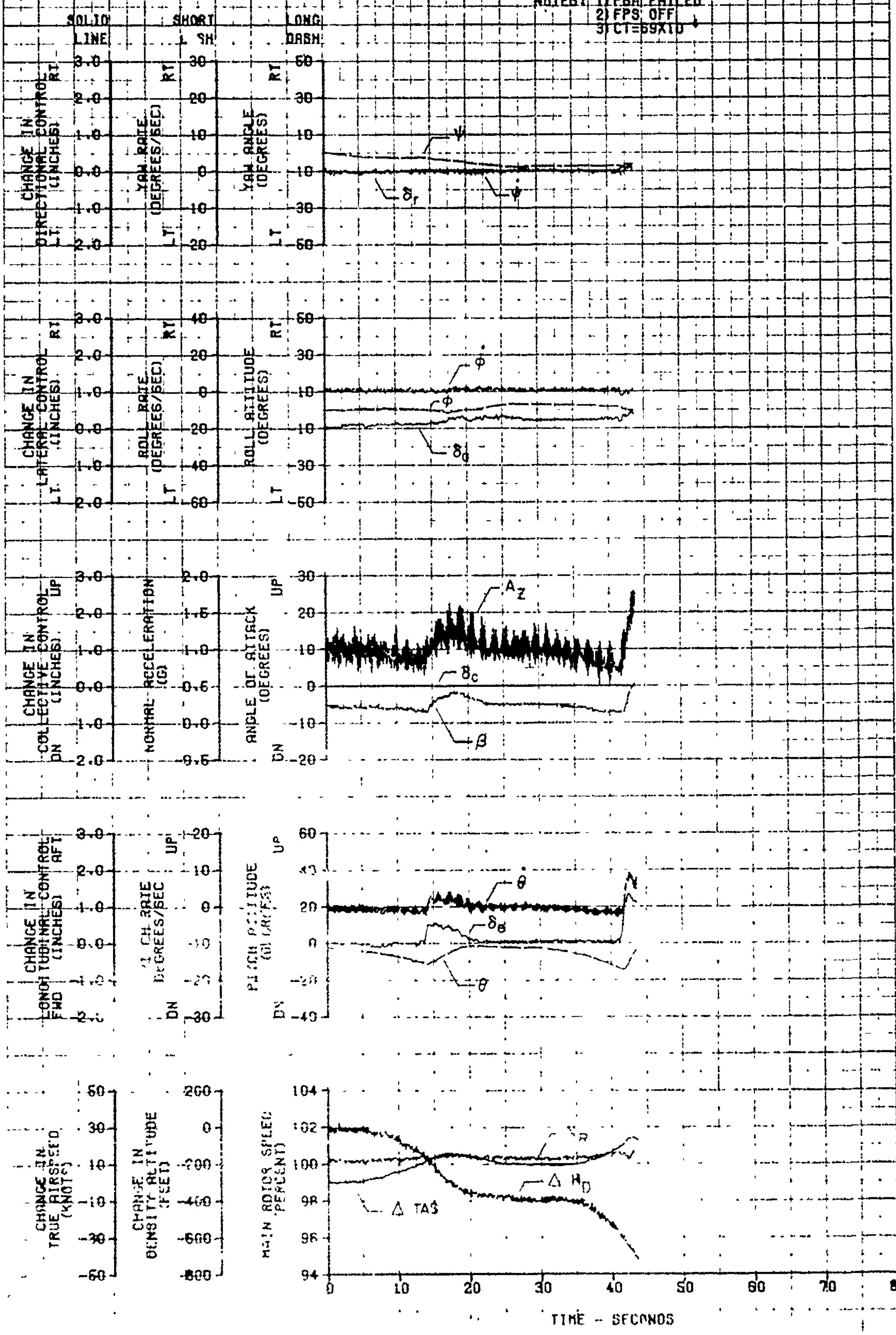


FIGURE 41
LONGITUDINAL LONG TERM RESPONSE
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
16200.	351.0(MID)	0.0(MID)	5220.	17.2	259.	91.	ON

NOTES: 1) PBA FAILED
2) FPS OFF
3) CT=59X10⁻⁴

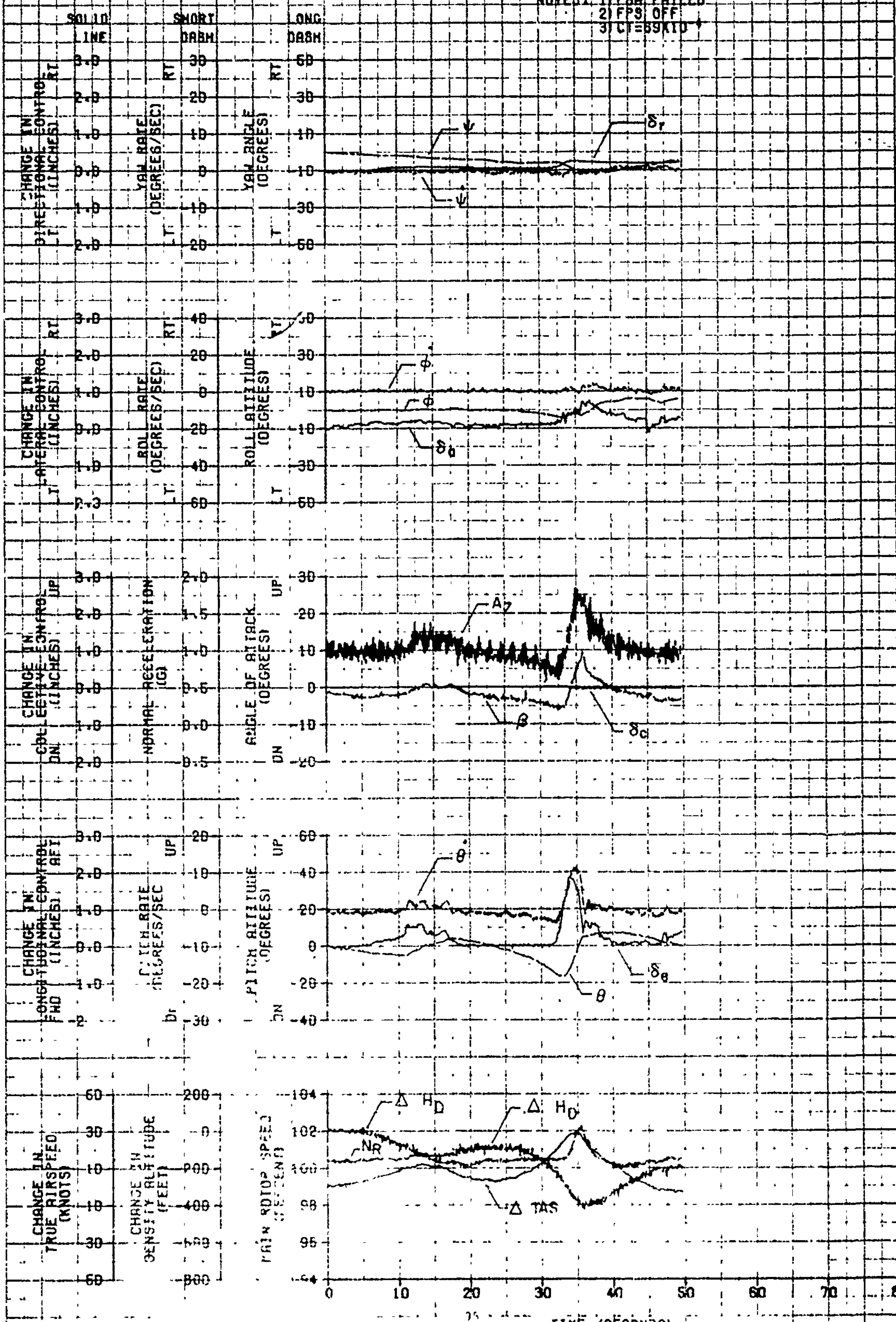
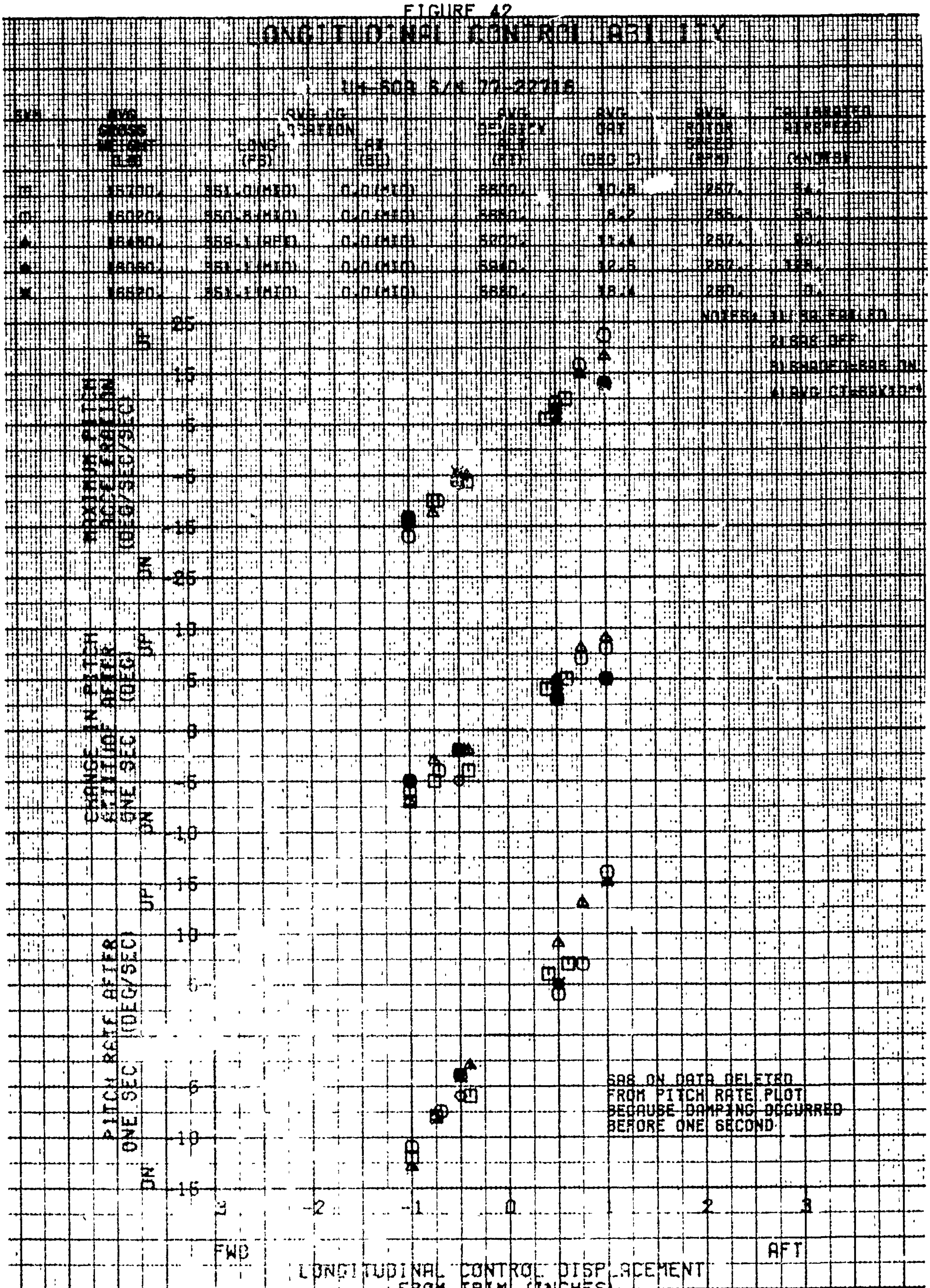
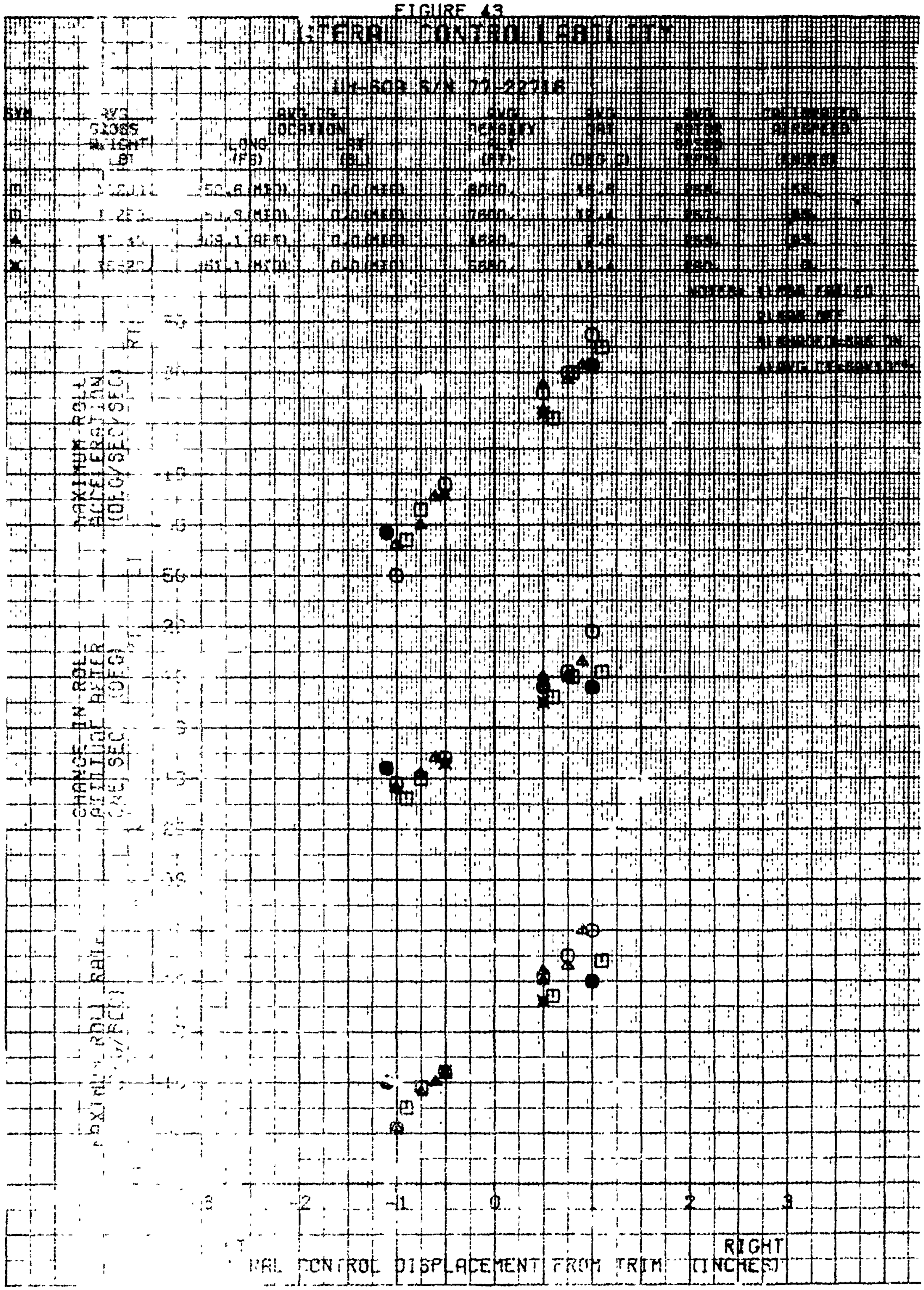


FIGURE 42



SAS ON DATA DELETED
FROM PITCH RATE PLOT
BECAUSE DAMPING OCCURRED
BEFORE ONE SECOND
PITCH RATE AFTER
ONE SEC (DEG/SEC)
FWD
AFT
LONGITUDINAL CONTROL DISPLACEMENT
FROM TRIM (INCHES)

FIGURE 43



LATERAL CONTROLLABILITY
UH-60A S/N 77-22716
SYM
GROSS WEIGHT (LB)
LONG (FS)
MAXIMUM ROLL ACCELERATION (DEG/SEC/SEC)
CHANGE IN ROLL ATTITUDE AFTER 1 SEC (DEG)
-2
-1
0
1
2
3
RIGHT
LATERAL CONTROL DISPLACEMENT FROM TRIM (INCHES)

FIGURE 44
DIRECTIONAL CONTROLLABILITY

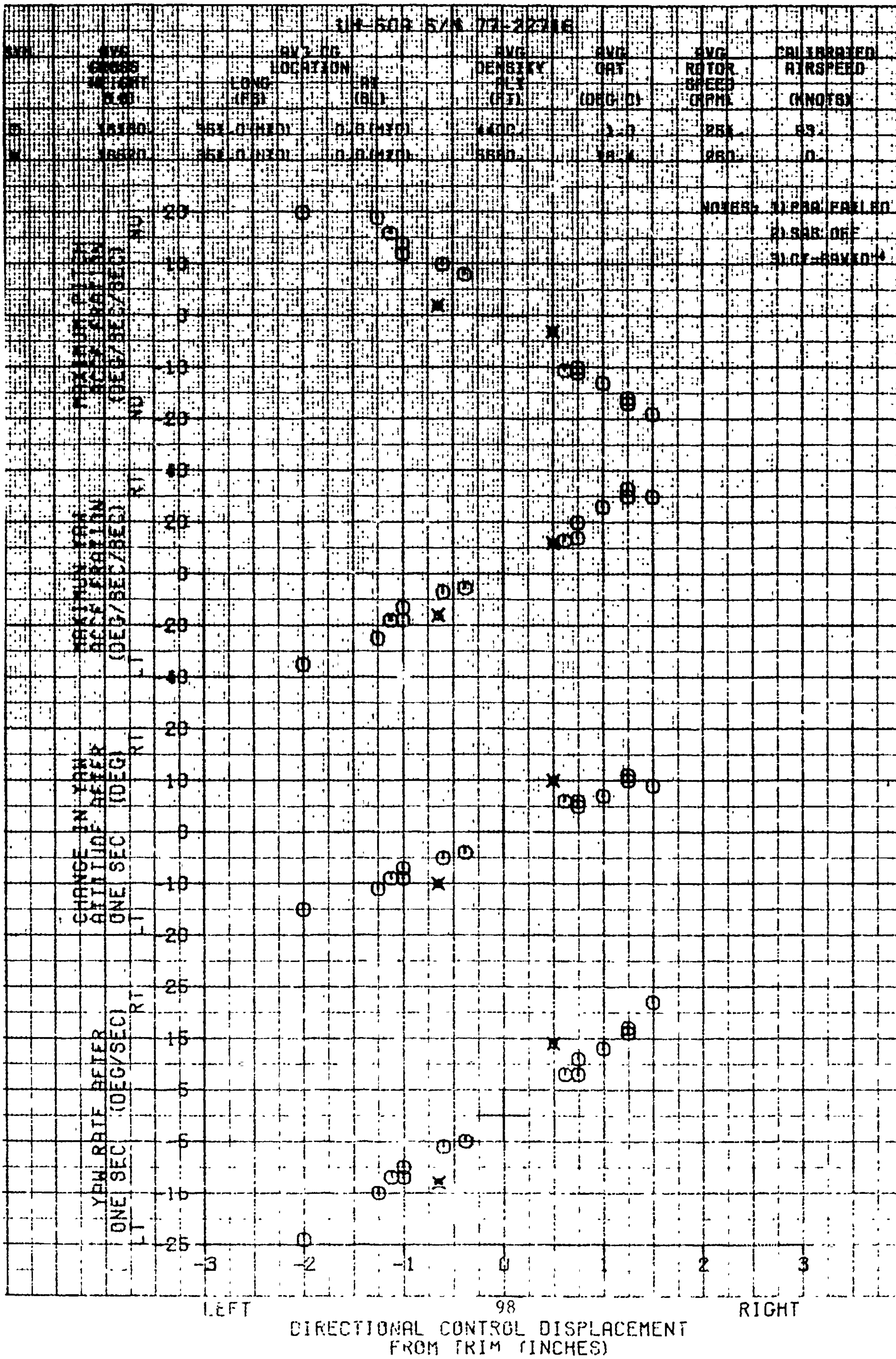
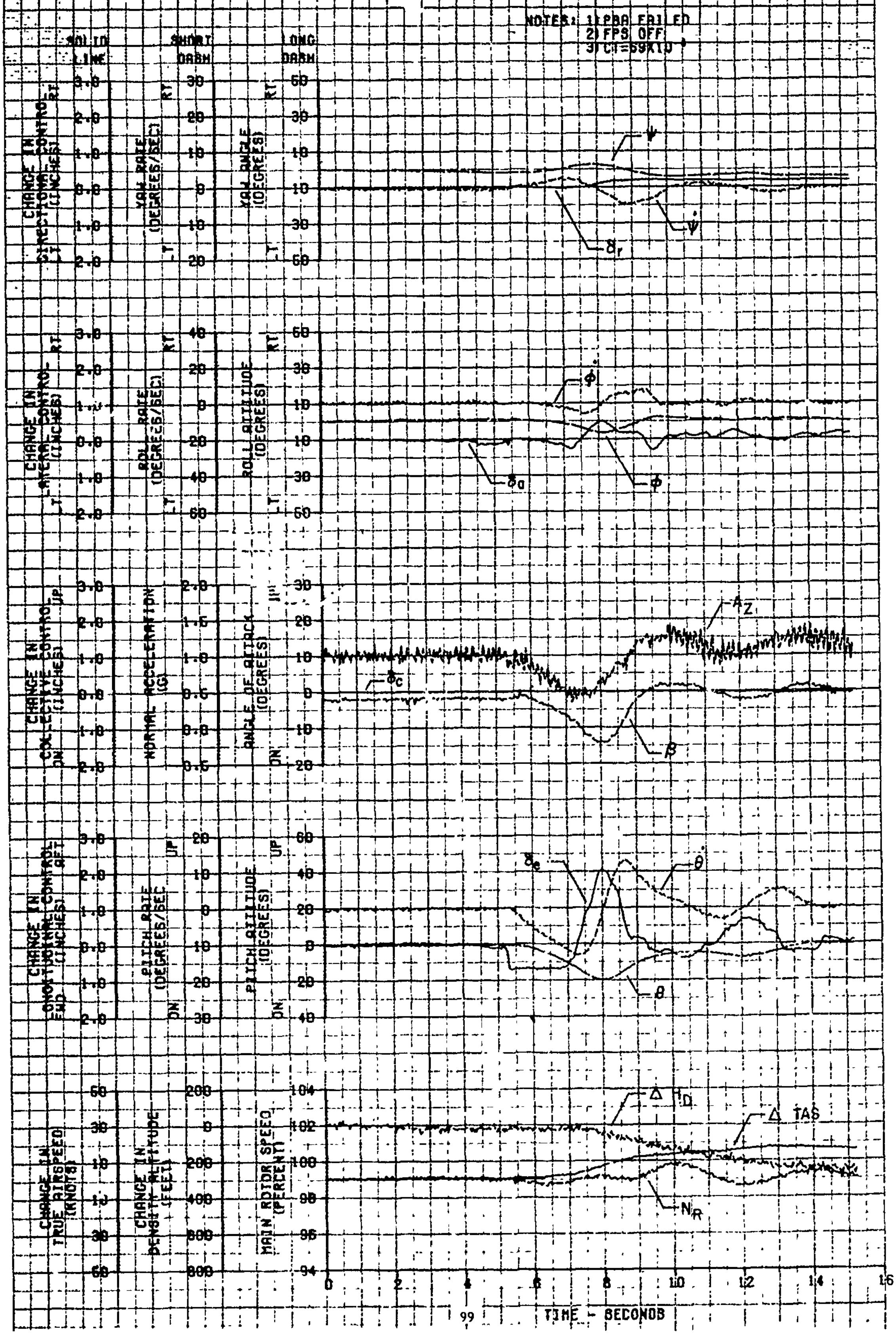


FIGURE 45
FORWARD LONGITUDINAL STEP INPUT
UH-60A USA S/N 77-22713

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
16020.	361.0(MID)	0.0(MID)	6700.	9.5	256.	91.	OFF



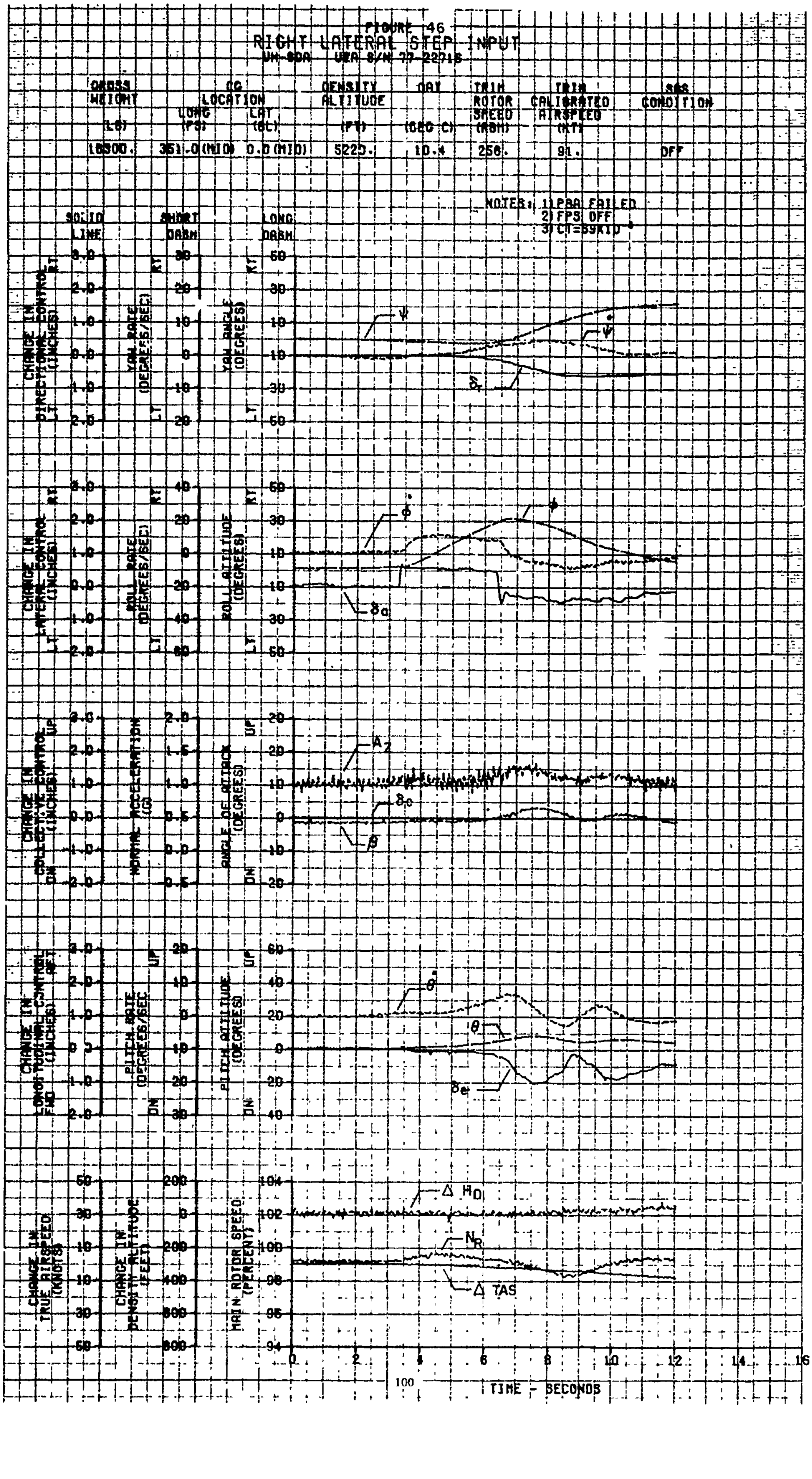


FIGURE 46
RIGHT LATERAL STEP INPUT
UH-60A USA S/N 77-22716

FIGURE 47
LEFT DIRECTIONAL STEP INPUT
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
16900.	351.0(MID)	0.0(MID)	4460.	5.0	253.	90.	OFF

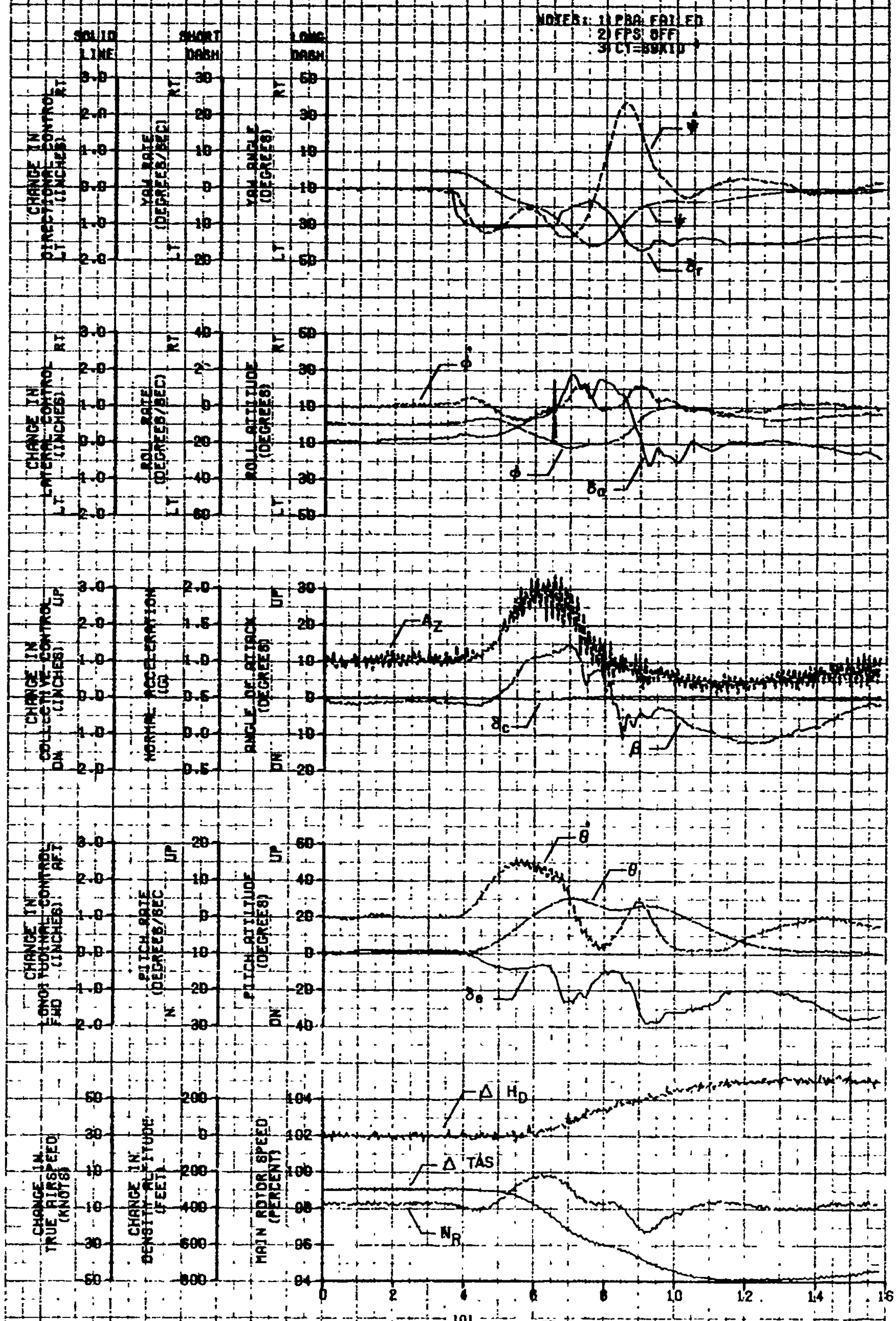


FIGURE 48
RIGHT DIRECTIONAL STEP INPUT
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
16520.	361.0 (MID)	0.0 (MID)	4140.	5.0	253.	91.	OFF

NOTES: 1) PBA FAILED
2) FPS OFF
3) CT=69X10

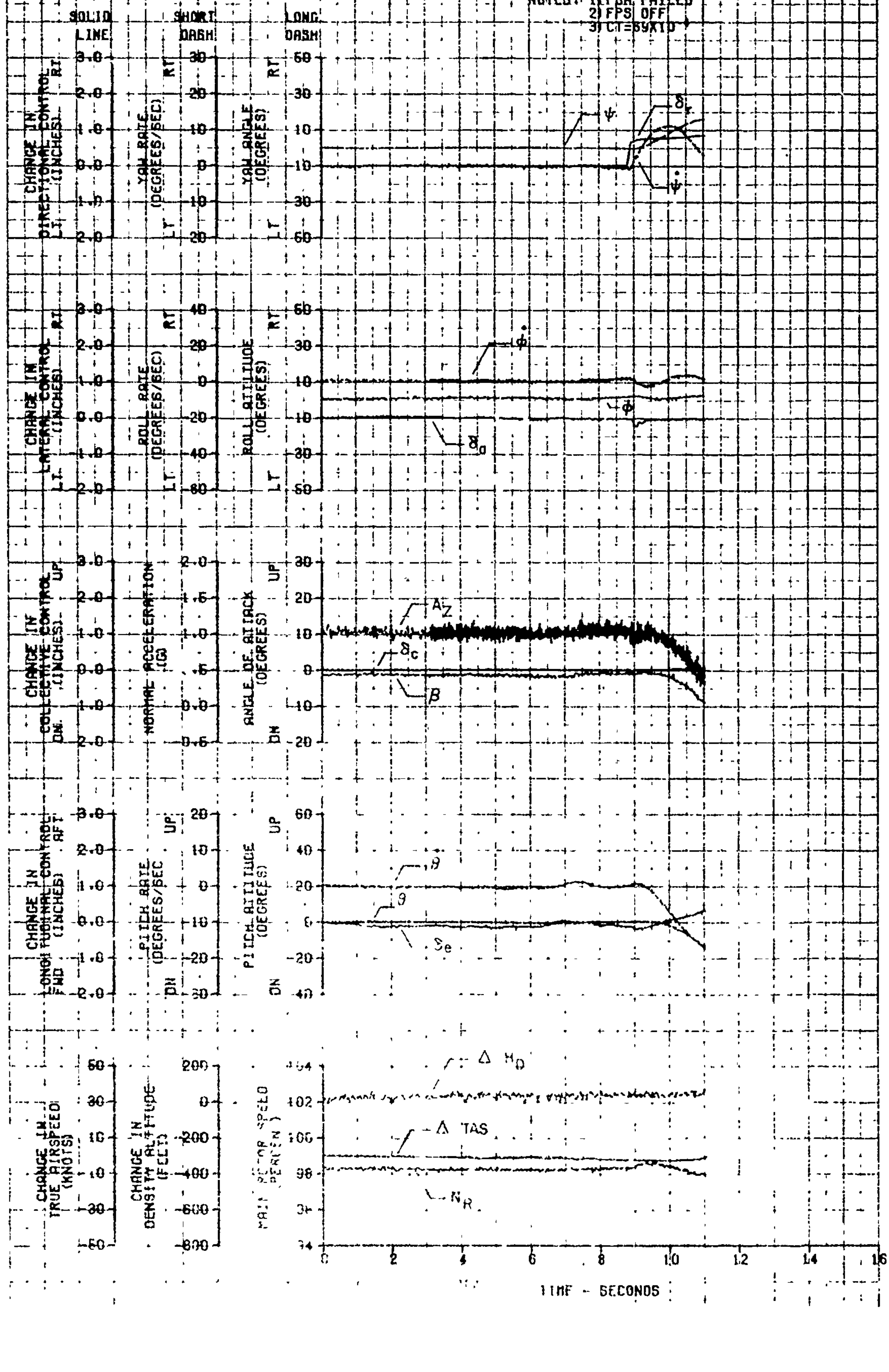


FIGURE 49

FWD-AFT LONGITUDINAL DOUBLET

UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
15540.	351.6(MID)	0.0(MID)	6420.	9.6	256.	91.	OFF

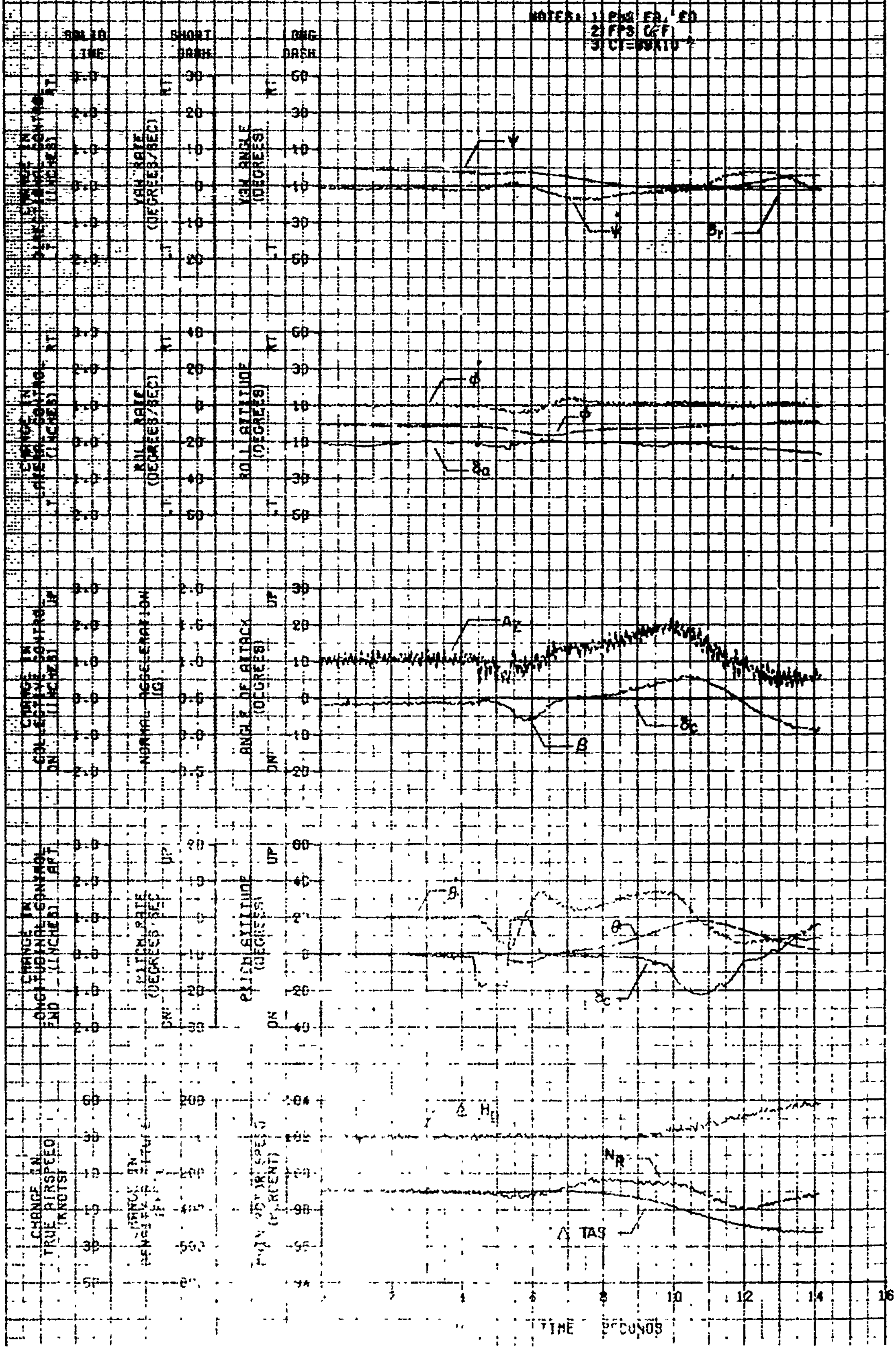


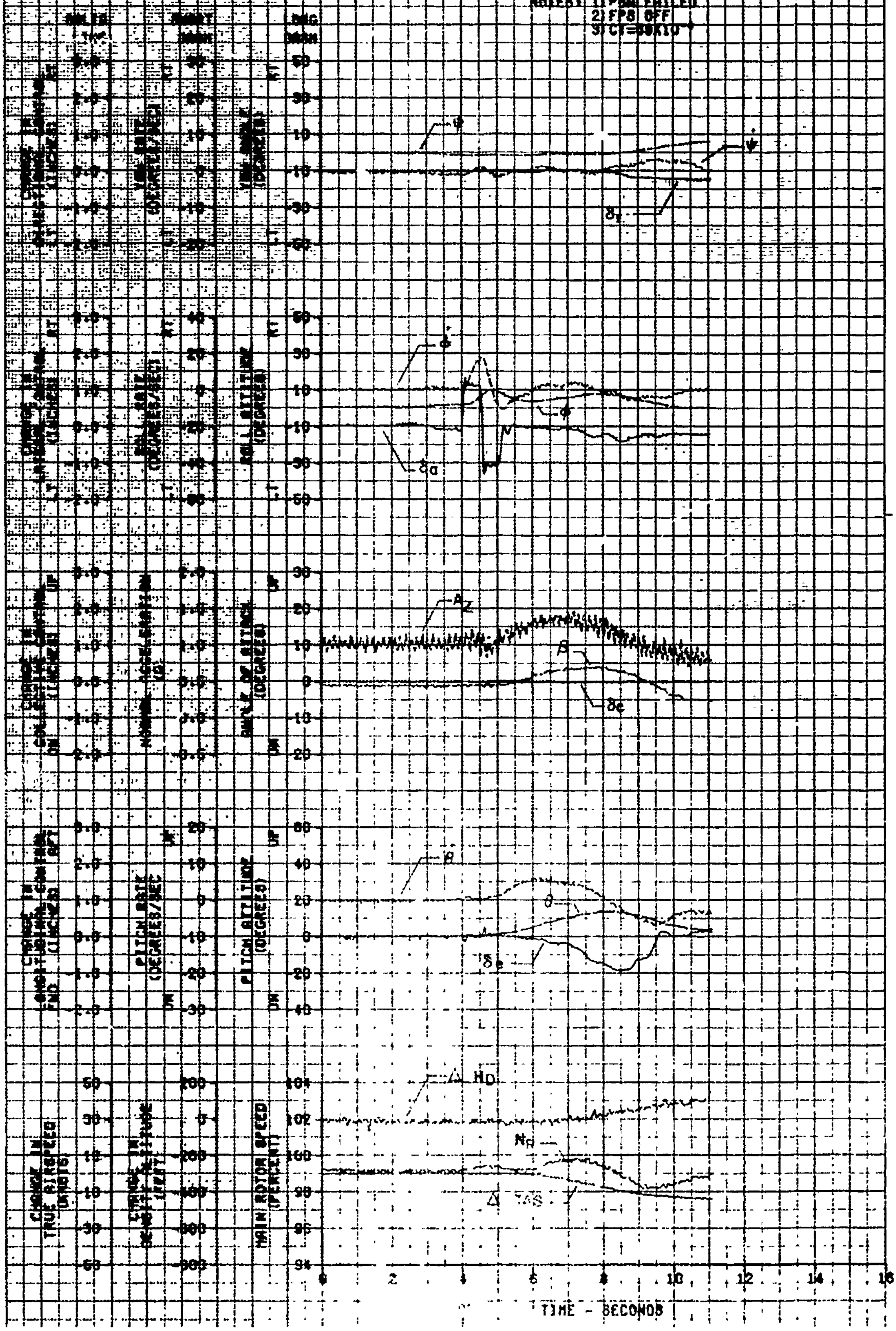
FIGURE 50

RIGHT-LEFT LATERAL DOUBLET

UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
[illegible]	360.0(MID)	0.0(MID)	5180.	10.2	258.	80.	OFF

NOTES: 1) PSA FAILED
2) FPS OFF
3) CT=[illegible]



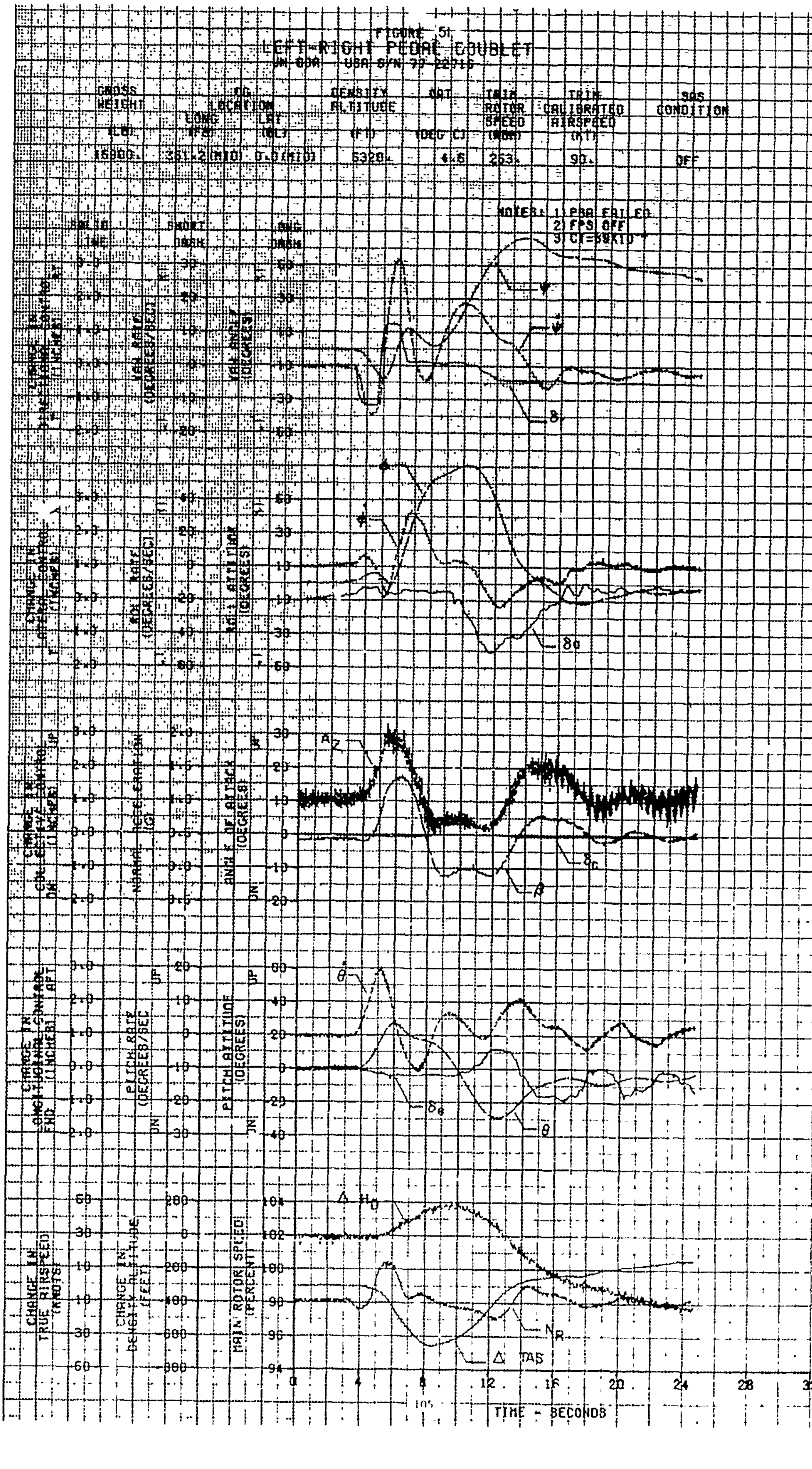


FIGURE 51
LEFT-RIGHT PEDAL DOUBLET
UH-60A USA S/N 77-22716
GROSS WEIGHT (LB) 16900.
CG LOCATION LONG (FS) 361.2(MID) LAT (BL) 0.0(MID)
DENSITY ALTITUDE (FT) 5320.
OAT (DEG C) 4.6
TRIM ROTOR SPEED (RPM) 253.
TRIM CALIBRATED AIRSPEED (KT) 90.
SAS CONDITION OFF
NOTES: 1) PBA FAILED
2) FPS OFF
SOLID LINE
SHORT DASH
LONG DASH
CHANGE IN DIRECTIONAL CONTROL (INCHES)
YAW RATE (DEGREES/SEC)
YAW ANGLE (DEGREES)
CHANGE IN LATERAL CONTROL (INCHES)
ROLL RATE (DEGREES/SEC)
ROLL ATTITUDE (DEGREES)
CHANGE IN COLLECTIVE CONTROL (INCHES)
NORMAL ACCELERATION (G)
ANGLE OF ATTACK (DEGREES)
CHANGE IN LONGITUDINAL CONTROL (INCHES)
PITCH RATE (DEGREES/SEC)
PITCH ATTITUDE (DEGREES)
CHANGE IN TRUE AIRSPEED (KNOTS)
CHANGE IN DENSITY ALTITUDE (FEET)
MAIN ROTOR SPEED (PERCENT)
TIME - SECONDS

FIGURE 52
RIGHT+LEFT PEDAL DOUBLET
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
16800.	361.0(MID)	0.0(MID)	5280.	2.2	252.	90.	OFF

NOTES: 1) PBA FAILED
2) FPS OFF
3) CT=89K10⁻⁴

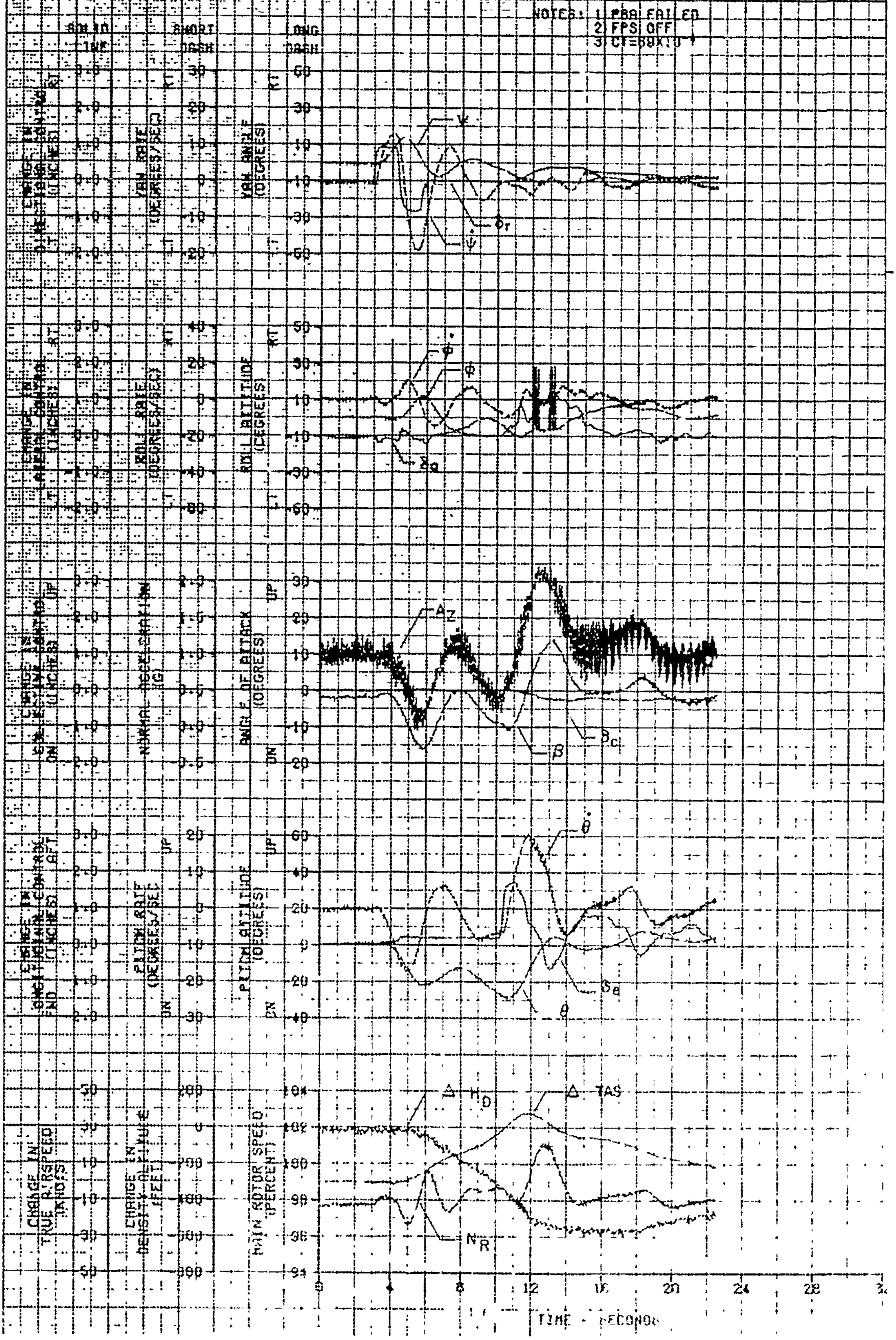


FIGURE 53
DOWN-UP COLLECTIVE DOUBLET
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
15860.	351.8(MID)	0.0(MID)	5380.	12.6	257.	54.	OFF

NOTES: 1) PBA FAILED
2) FPS OFF
3) CT=.0089

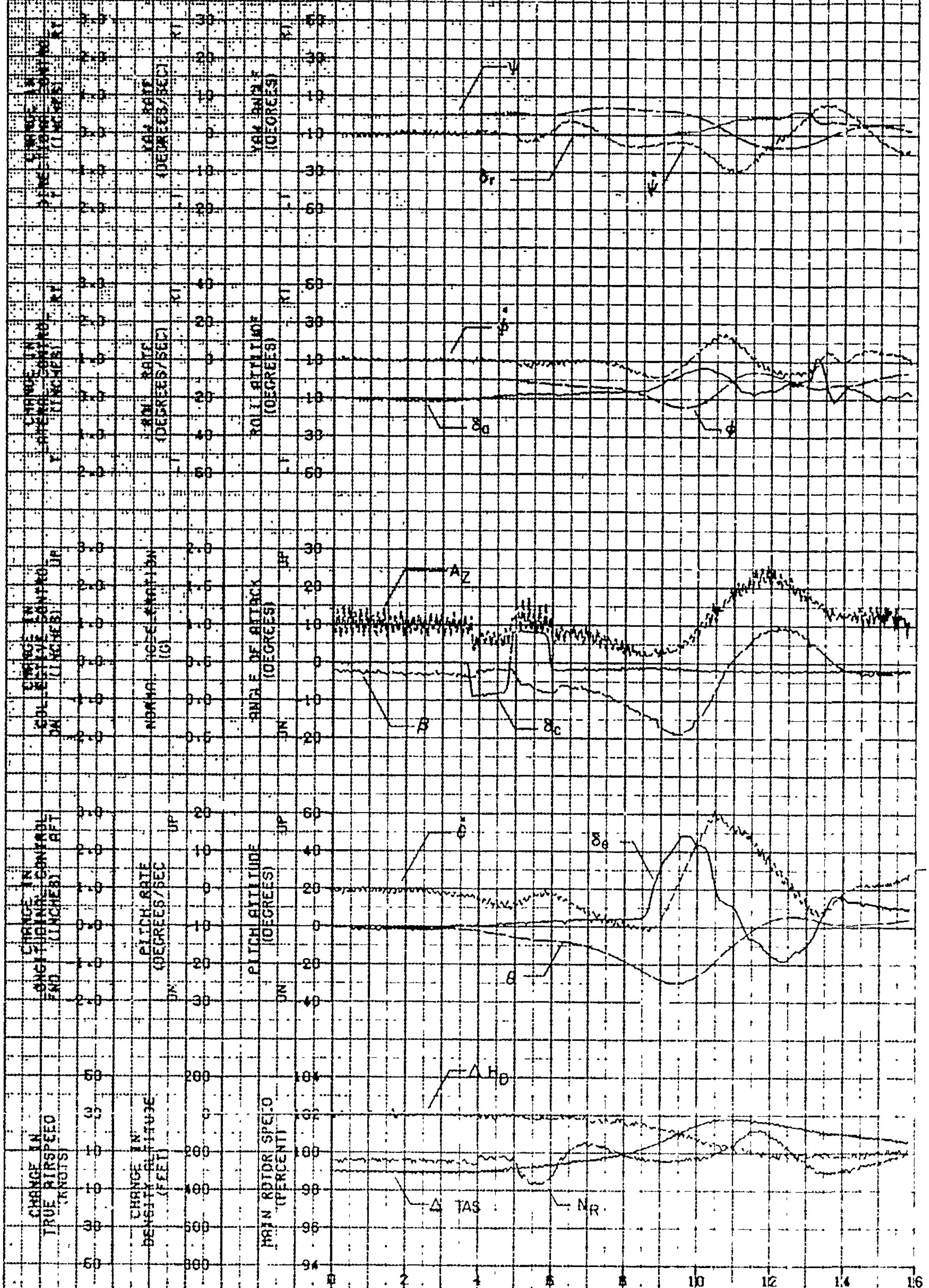
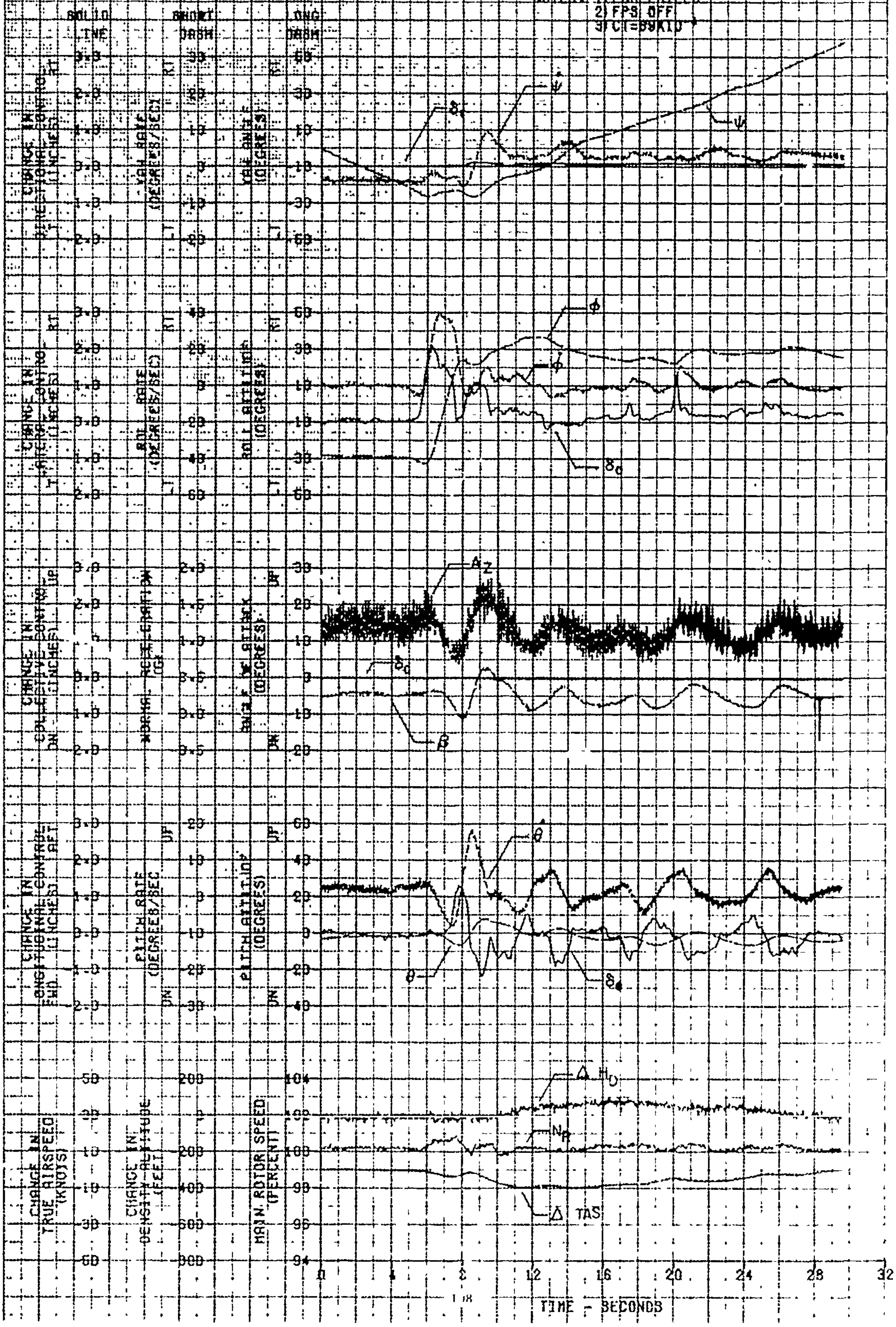


FIGURE 54
LEFT-TO-RIGHT ROLL REVERSAL
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
16740.	351.6(MID)	0.0(MID)	7080.	16.6	259.	125.	OFF

NOTES: 1) PBA FAILED
2) FPS OFF
3) CT=69KID



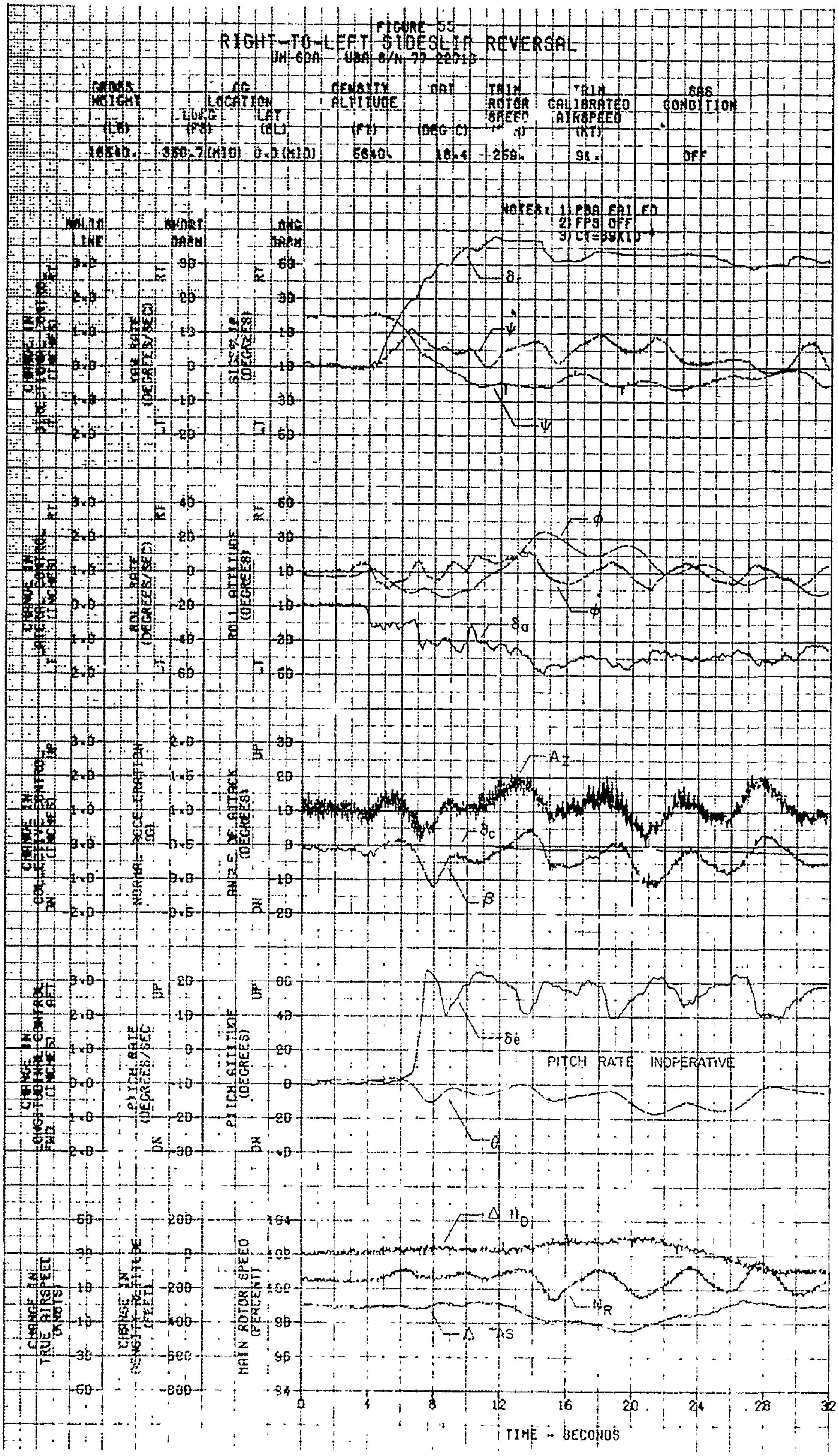


FIGURE 55
RIGHT-TO-LEFT SIDESLIP REVERSAL
GROSS WEIGHT (LB)
CG LOCATION
LONG (FS)
LAT (BL)
DENSITY ALTITUDE (FT)
OAT (DEG C)
TRIM ROTOR SPEED
TRIM CALIBRATED AIRSPEED (KT)
SAS CONDITION
350.7 (MID)
0.0 (MID)
18.4
258.
91.
OFF
NOTES: 1) PBA FAILED
2) FPS OFF
SOLID LINE
SHORT DASH
LONG DASH
PITCH RATE INOPERATIVE
TIME - SECONDS

FIGURE 56
LEFT-TO-RIGHT SIDESLIP REVERSAL
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
16520.	350.7(MID)	0.0(MID)	5820.	17.6	258.	92.	OFF

NOTES: 1) PBA FAILED
2) FPS OFF
3) CT=89X10⁻⁴

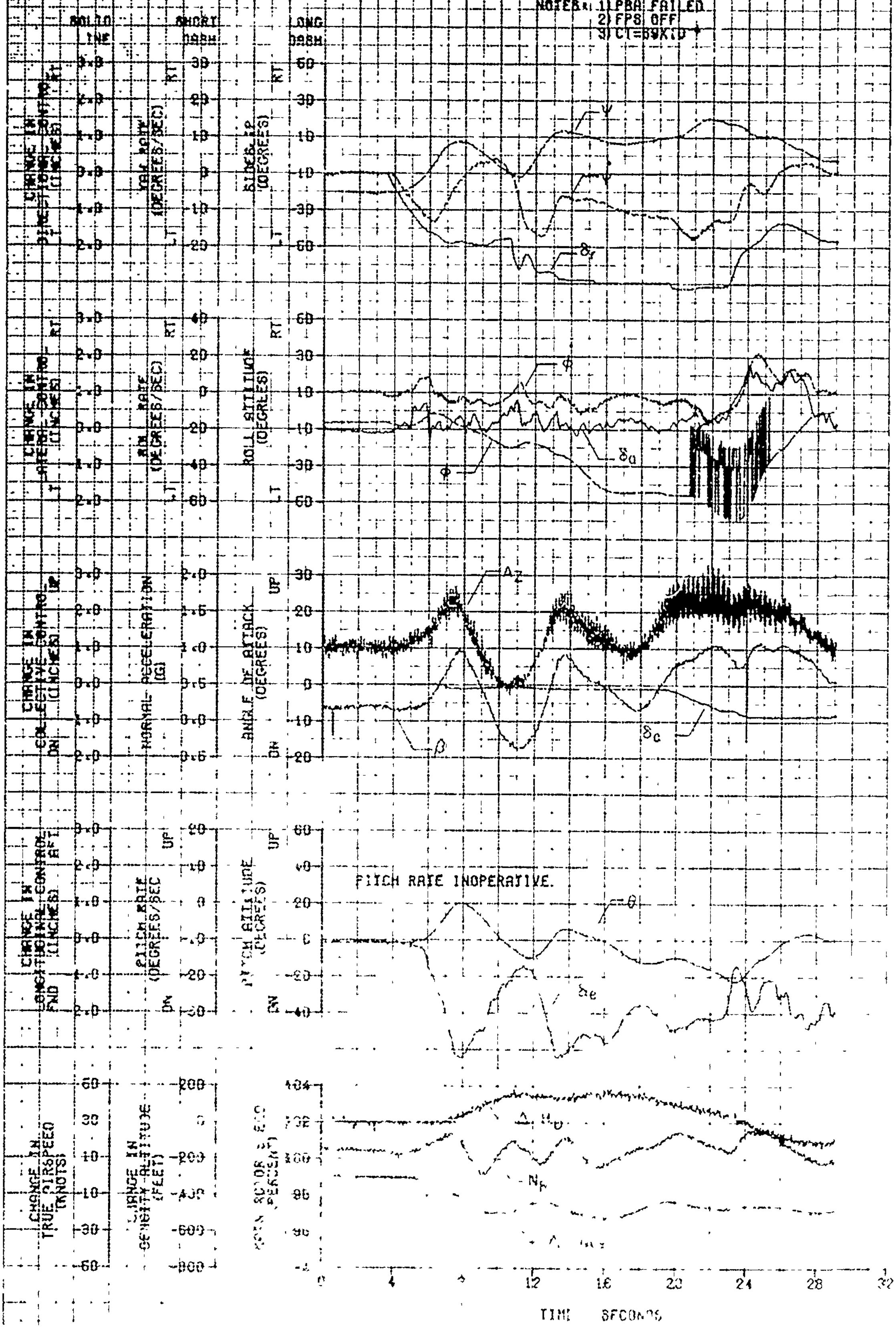


FIGURE 57
RIGHT-TO-LEFT SIDESLIP REVERSAL
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
16480.	360.9(MID)	0.0(MID)	5700.	17.8	258.	92.	OFF

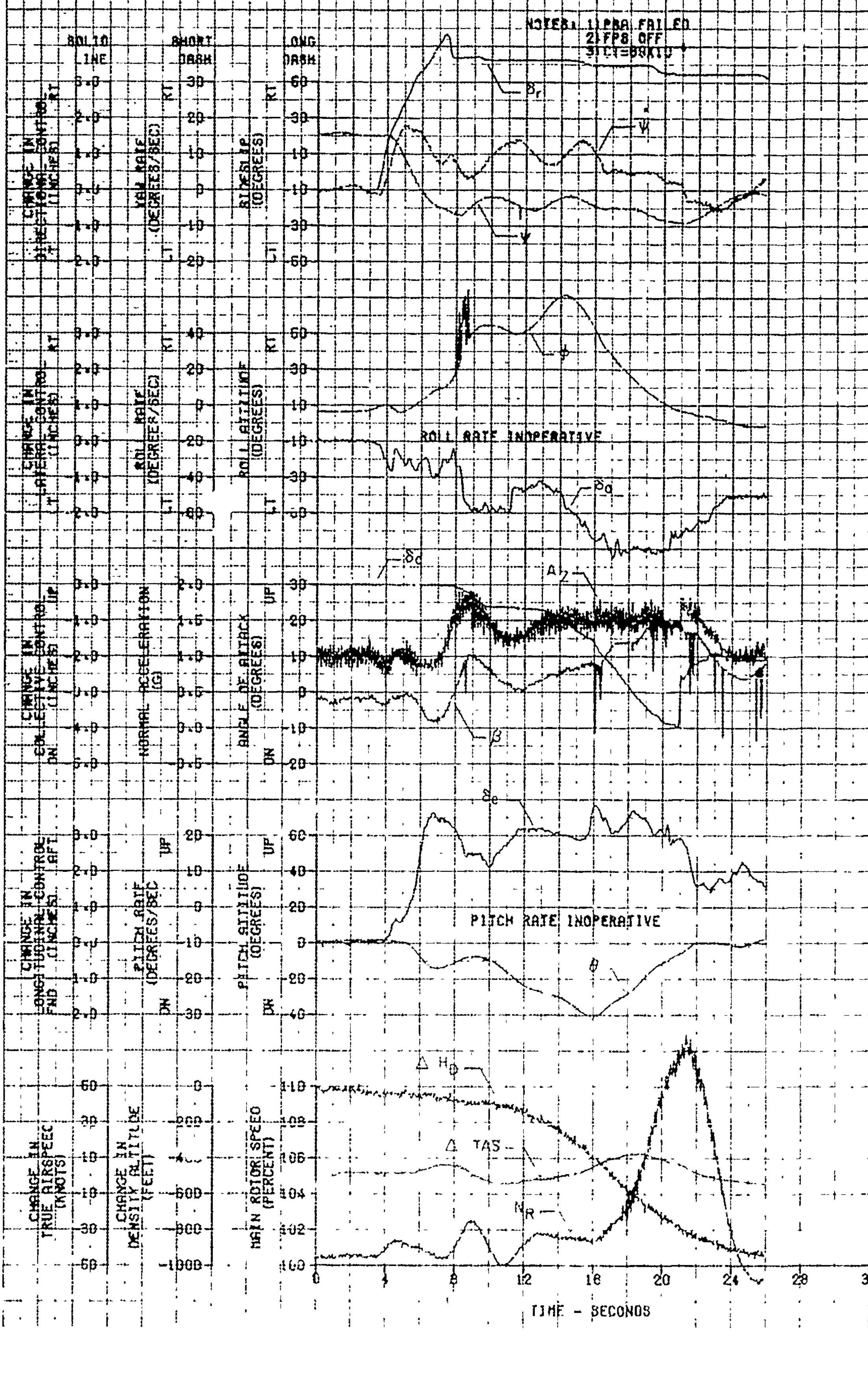
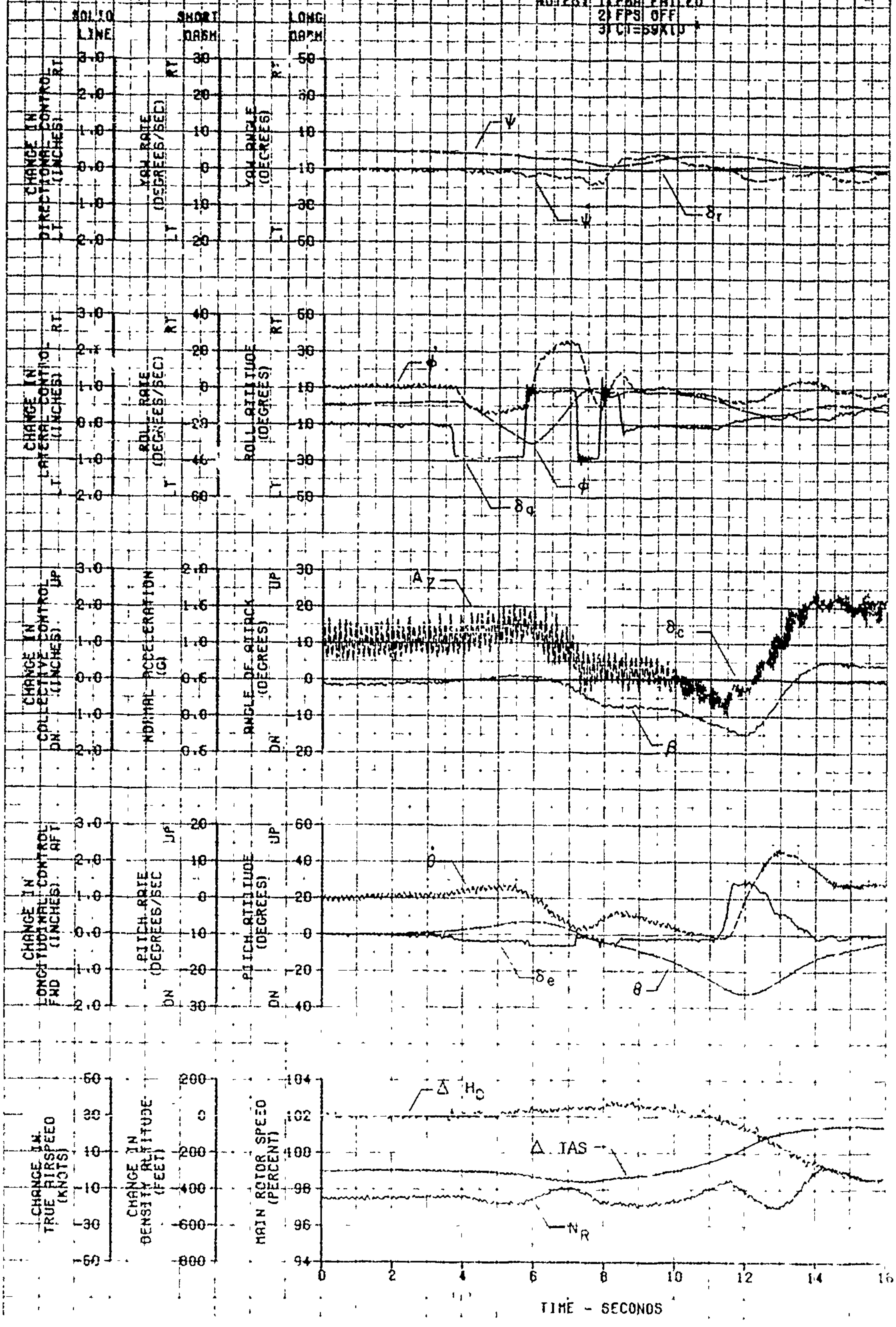


FIGURE 58
LEFT LATERAL S.I. MANEUVER
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
15880.	351.0 (MID)	0.0 (MID)	4920.	0.4	251.	91.	OFF

NOTES: 1) PBA FAILED
2) FPS OFF
3) CT=59KID



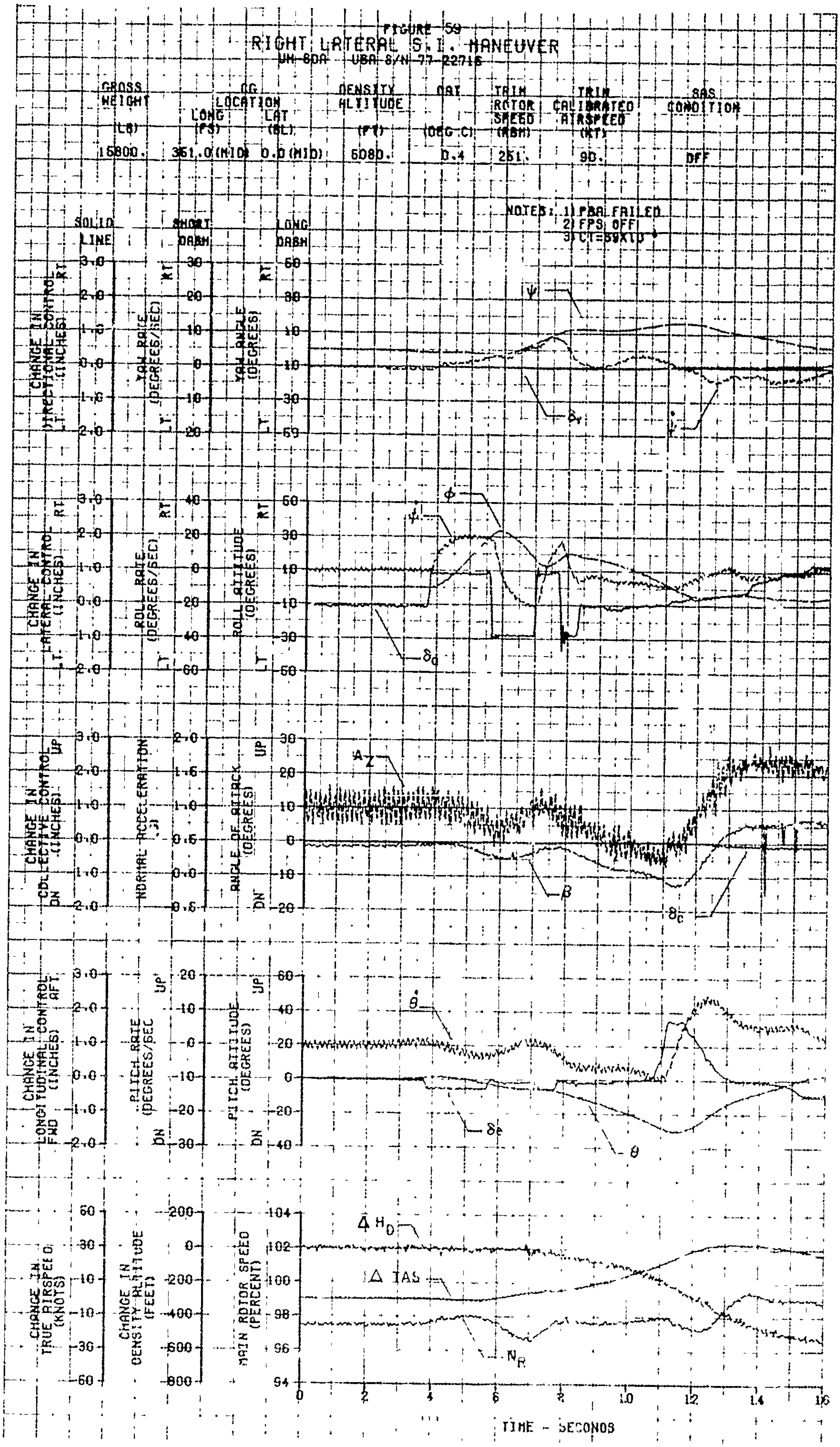


FIGURE 59
RIGHT LATERAL S.I. MANEUVER
UH-60A USA S/N 77-22716

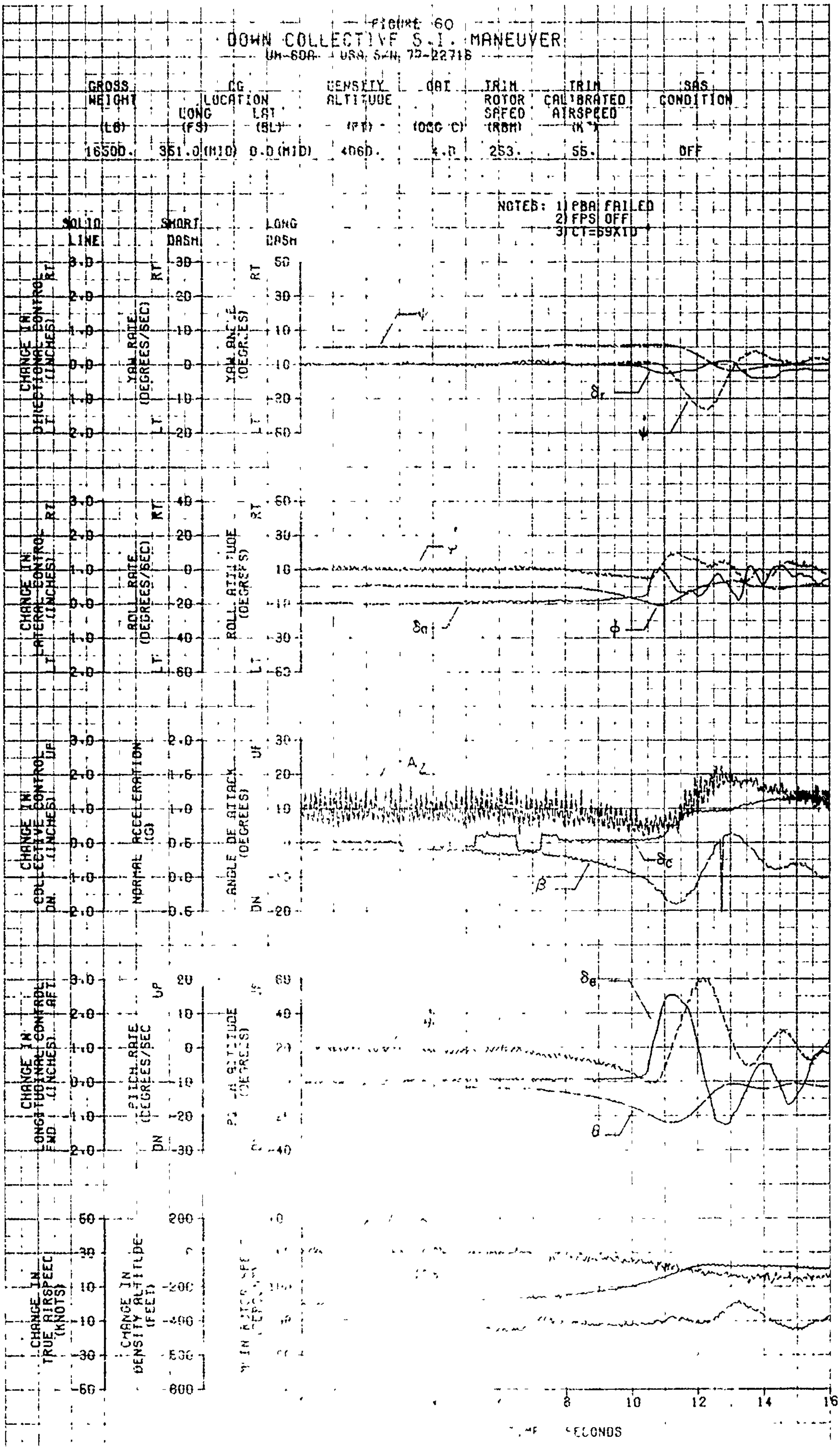


FIGURE 60
DOWN COLLECTIVE S.I. MANEUVER
UH-60A USA S/N 77-22716
GROSS WEIGHT (LB) 16500.
CG LOCATION LONG (FS) 361.0 (MID) LAT (BL) 0.0 (MID)
DENSITY ALTITUDE (FT) 4060.
OAT (DEG C)
TRIM ROTOR SPEED (RPM) 253.
TRIM CALIBRATED AIRSPEED (KT) 55.
SAS CONDITION OFF
NOTES: 1) PBA FAILED
2) FPS OFF
3) CT=69X10
SOLID LINE
SHORT DASH
LONG DASH
CHANGE IN DIRECTIONAL CONTROL (INCHES)
YAW RATE (DEGREES/SEC)
YAW ANGLE (DEGREES)
CHANGE IN LATERAL CONTROL (INCHES)
ROLL RATE (DEGREES/SEC)
ROLL ATTITUDE (DEGREES)
CHANGE IN COLLECTIVE CONTROL (INCHES)
NORMAL ACCELERATION (G)
ANGLE OF ATTACK (DEGREES)
CHANGE IN LONGITUDINAL CONTROL (INCHES)
PITCH RATE (DEGREES/SEC)
PITCH ATTITUDE (DEGREES)
CHANGE IN TRUE AIRSPEED (KNOTS)
CHANGE IN DENSITY ALTITUDE (FEET)
RT
LT
UP
DN
AFT
FWD
ψ
δ_r
ϕ
δ_a
A_Z
δ_c
β
δ_e
θ
8
10
12
14
16
TIME SECONDS

FIGURE 61
UP COLLECTIVE S.I. MANEUVER
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
16520.	361.0 (MID)	0.0 (MID)	4080.	4.6	253.	55	OFF

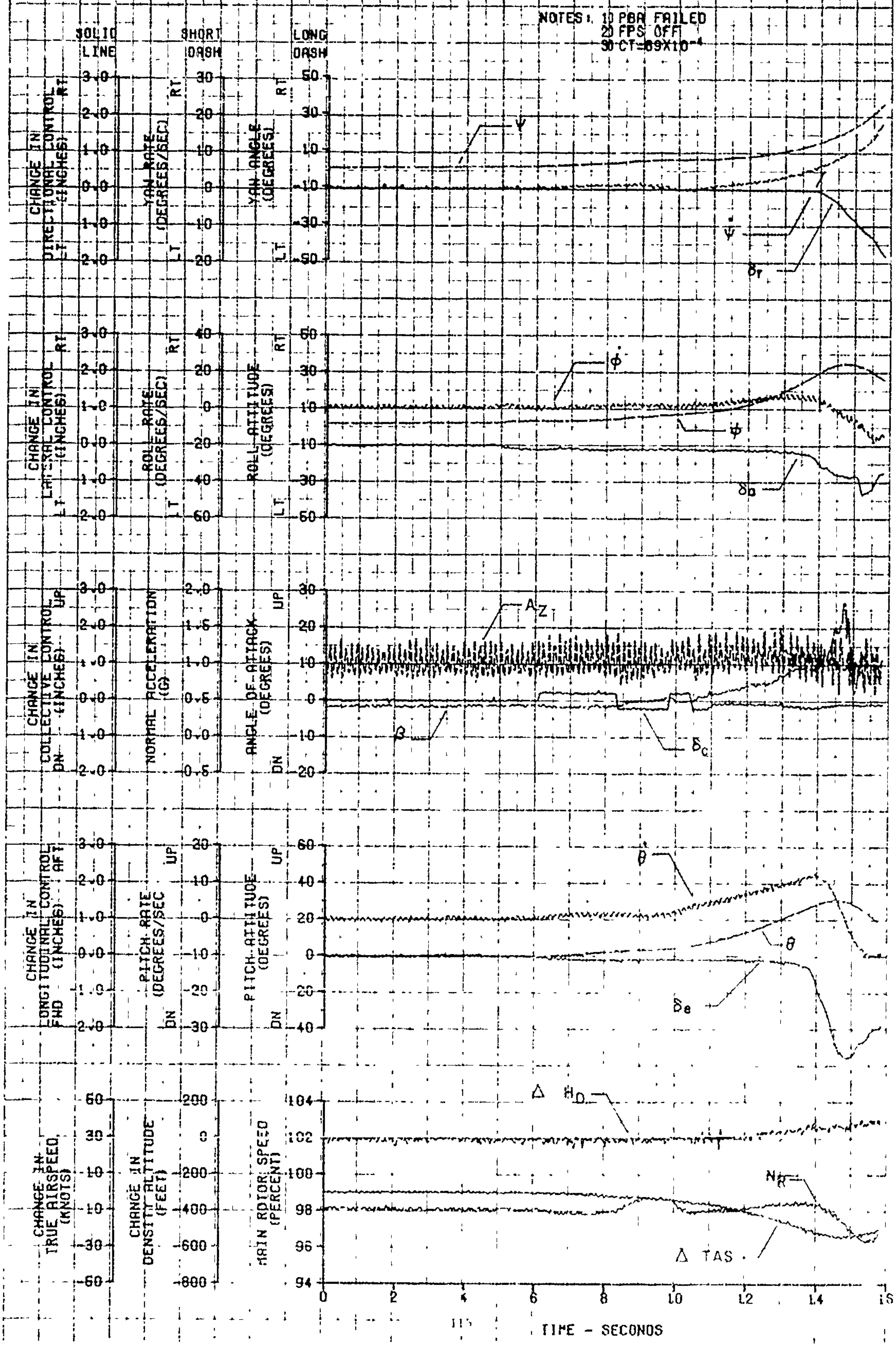
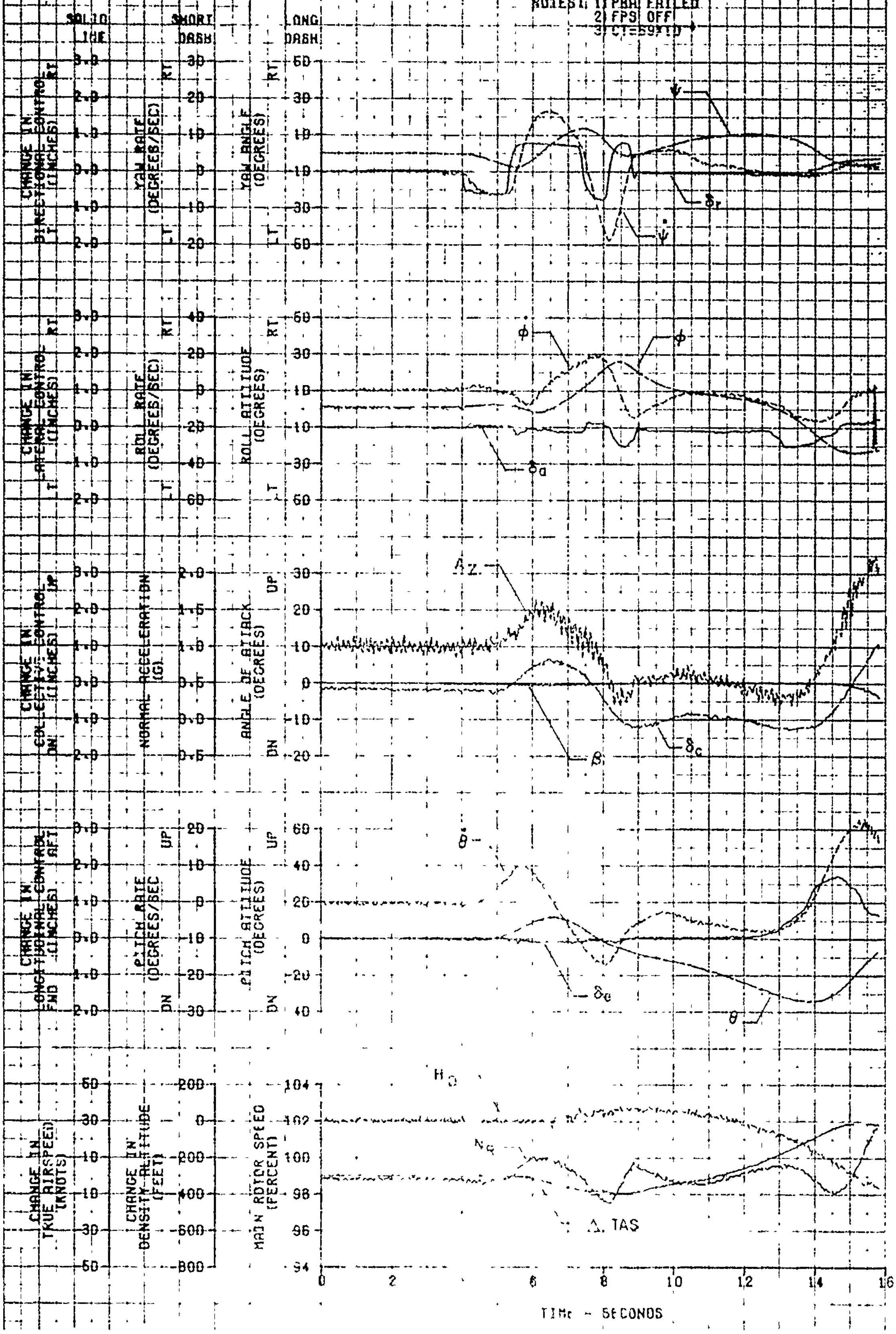


FIGURE 62
RIGHT DIRECTIONAL S.I. MANEUVER
UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
15720.	351.0(MID)	0.0(MID)	6200.	0.6	255.	90.	OFF

NOTES: 1) PBA FAILED
2) FPS OFF
3) CT=59X10



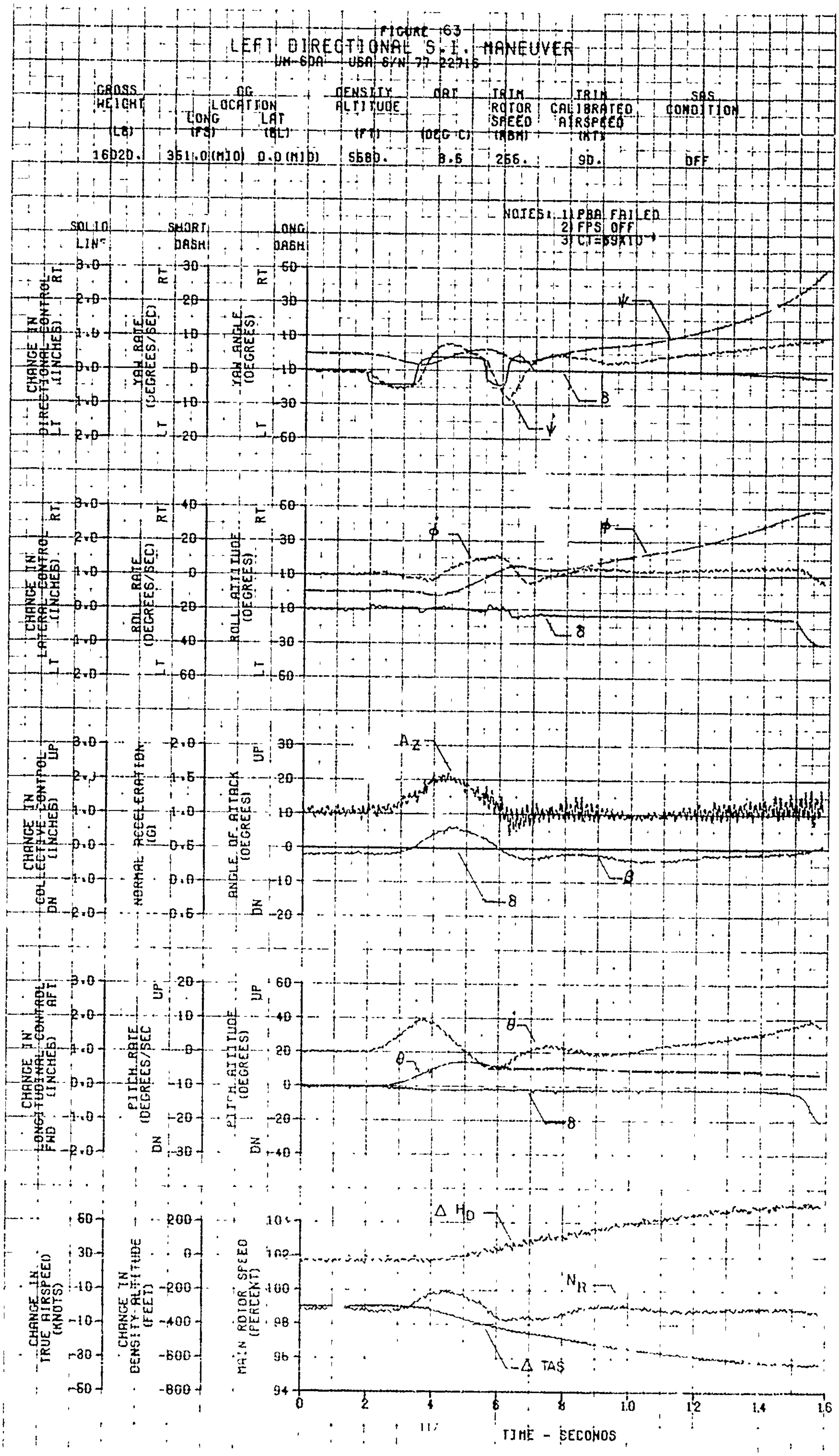


FIGURE 63
LEFT DIRECTIONAL S.I. MANEUVER
UH-60A USA S/N 77-22716
GROSS WEIGHT (LB) 16020.
CG LOCATION LONG (FS) 351.0(MID) LAT (BL) 0.0(MID)
DENSITY ALTITUDE (FT) 5580.
OAT (DEG C) 8.6
TRIM ROTOR SPEED (RPM) 256.
TRIM CALIBRATED AIRSPEED (KT) 90.
SAS CONDITION OFF
NOTES: 1) PBA FAILED
2) FPS OFF
3) CT=59X10⁻⁴
SOLID LINE
SHORT DASH
LONG DASH
CHANGE IN DIRECTIONAL CONTROL (INCHES)
YAW RATE (DEGREES/SEC)
YAW ANGLE (DEGREES)
CHANGE IN LATERAL CONTROL (INCHES)
ROLL RATE (DEGREES/SEC)
ROLL ATTITUDE (DEGREES)
CHANGE IN COLLECTIVE CONTROL (INCHES)
NORMAL ACCELERATION (G)
ANGLE OF ATTACK (DEGREES)
CHANGE IN LONGITUDINAL CONTROL (INCHES)
PITCH RATE (DEGREES/SEC)
PITCH ATTITUDE (DEGREES)
CHANGE IN TRUE AIRSPEED (KNOTS)
CHANGE IN DENSITY ALTITUDE (FEET)
MAIN ROTOR SPEED (PERCENT)
RT
LT
UP
DN
AFT
FWD
ψ
β
δ
ϕ
A_Z
θ
ΔH_D
N_R
Δ TAS
TIME - SECONDS
0
2
4
6
8
10
12
14
16

FIGURE 64

FORWARD LONGITUDINAL S.I. MANEUVER

UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
16200.	361.0(MID)	0.0(MID)	5660.	19.6	257.	64.	OFF

NOTES: 1) PBA FAILED
2) FPS OFF
3) CT=89XID

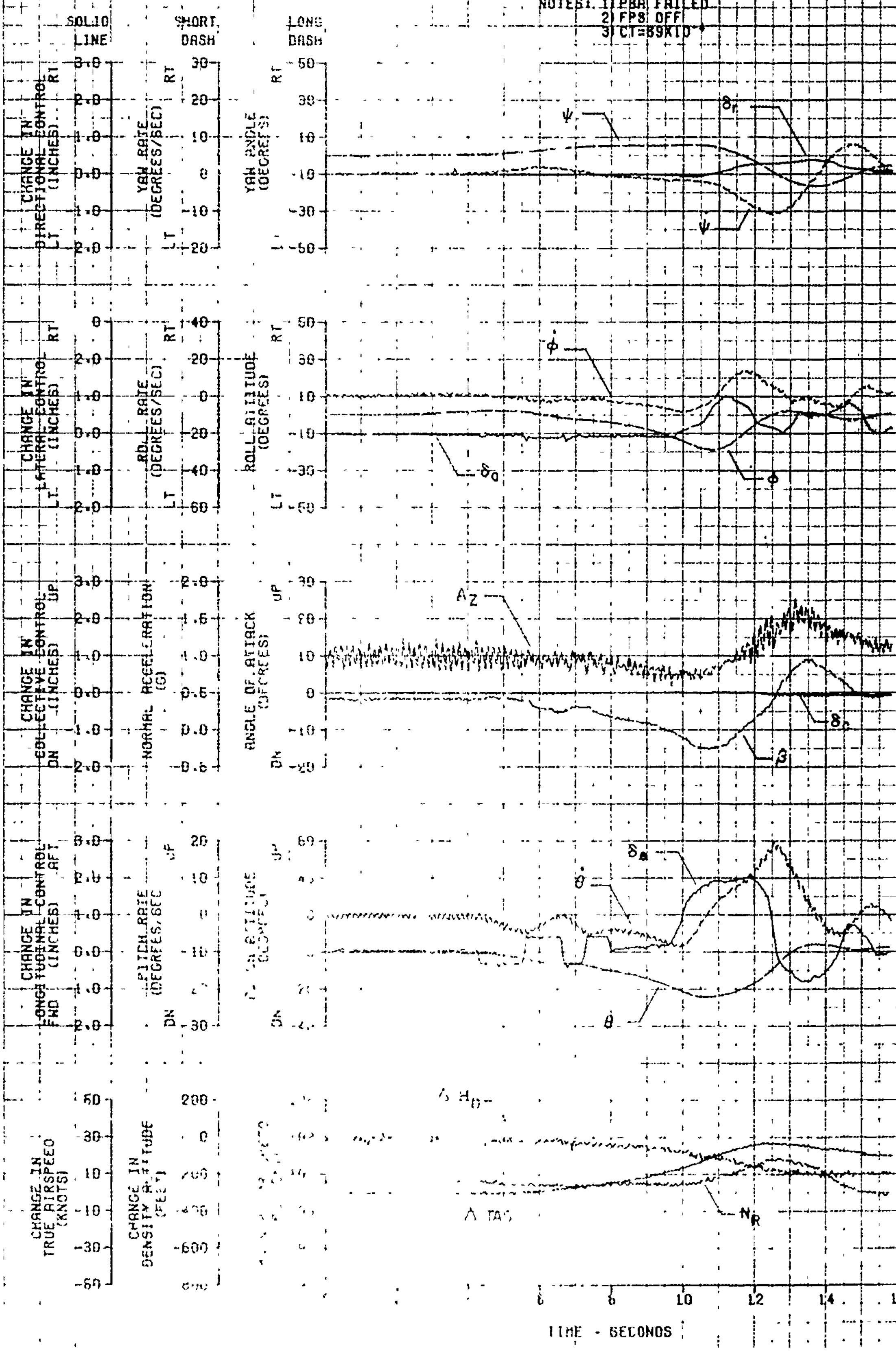


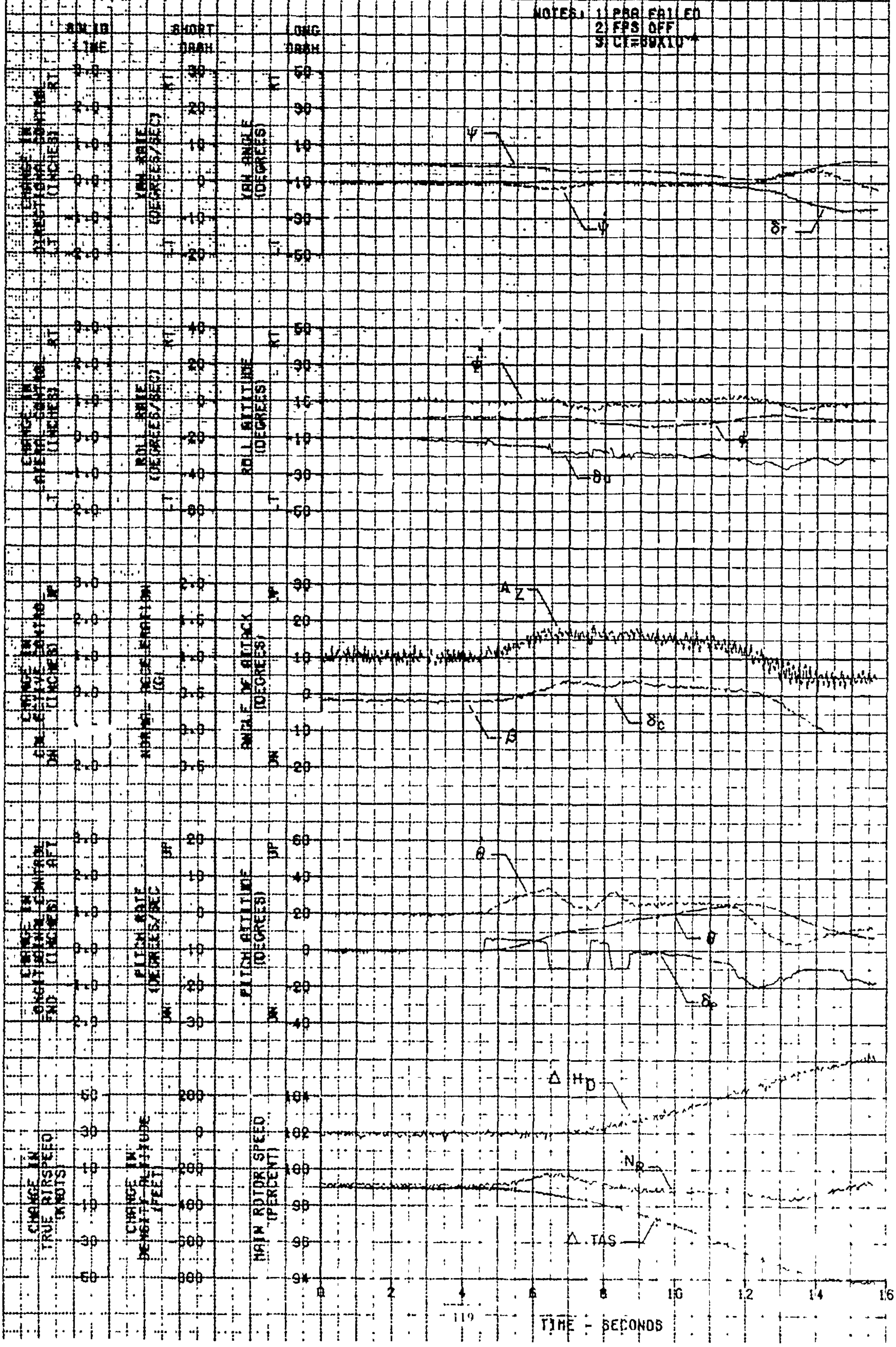
FIGURE 65

AFT LONGITUDINAL S.I. MANEUVER

UH-60A USA S/N 77-22716

GROSS WEIGHT (LB)	CG LOCATION LONG (FS)	CG LOCATION LAT (BL)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SAS CONDITION
15860.	361.0(MID)	0.0(MID)	6060.	10.0	256.	91.	OFF

NOTES: 1) PBA FAILED
2) FPS OFF
3) C_T=89X10^{-4}



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